1997

Martian communications network design trade study

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Martian Communications Network

Design Trade Study

A Thesis

Presented to

The Faculty of the Department of Mechanical and Aerospace Engineering

San Jose State University

In Partial Fulfillment

of the Requirements for the Degree

Master of Science

by

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December, 1997

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Abstract

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All spacecraft missions require some form of communication subsystem to relay data to a mission control point. With the luxury of an established communication network for Mars exploration, future Mars exploration/colonization programs could readily afford a larger payload, or use a less powerful, less expensive launcher to send their equipment to Mars. This work examines several satellite constellations to determine their suitability for acting as Relay satellites for a Mars Communications Network.

Several scenarios for the configuration of a network constellation, based on present technology and past publications for communication network designs, are outlined. Orbital parameters are determined for each constellation. Based upon these orbits, several properties of each constellation, such as the communications coverage, data rates, and spacecraft configurations, are determined. Finally, estimates of which constellation best suits users' needs are made.

Acknowledgments

To my wife, Lorraine, for putting up with my seven-year journey toward this degree, and to my darling, somewhat patient daughters, Karley and Shawna and little Keira, and baby son Deven, for understanding that their Papa had homework to do, too. I would also like to thank Drs. Desautel and Marouf for their time and effort advising me on this work, and Dr Pernicka, for four and a half years of good advice.

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CHAPTER 1
Introduction

From the time humans discovered that the reddish star in the heavens was actually a neighboring planet, we have wondered what Mars is like. What caused the features that we see today, those that made our ancestors believe that there was life on Mars? What caused its atmosphere to thin, to "leak away?" Could it happen here on Earth? Can humans live there? The first steps in answering these question will be sending robotic probes to the Red Planet, likely followed by human exploration. All of these missions will require communications with Earth. In the past, each mission has used a dedicated orbiter for relaying communications to/from Earth. Sometimes these orbiters carry additional sensors, used to gather data about the planet. In all cases, large sums of money are spent to relay that data to scientists here on Earth. What if some of those costs could be spread across several missions? A portion of the mass dedicated to communications could be removed from each of these missions, allowing the use of a smaller, less costly launch vehicle to send the mission on its way, or the same launcher could send a much greater quantity of scientific instruments to Mars, allowing scientists to gather more data. Either way, the benefits would be enormous.

This trade study examines several satellite constellation configurations for an Earth - Mars Communications Network. The work was divided into two parts and between two team members: the Earth - Mars transfer orbit analysis, constellation deployment time, parametric cost equation development, and mission monetary cost[1]; and the communications analysis, the derivation/integration of constellation satellite orbits about Mars, the communications coverage analysis, and spacecraft configuration (this work). Each configuration is scored in several areas, such as monetary cost, communications availability, time to establish the network, etc. These results are then used as the input to a parametric cost equation, of the form

$$J=\sum_{i=1}^{n}(A_i * a_i)$$

where J represents the total "cost", a_i represents the parametric portion of each term, and A_i represents the weighting factor for each term.

Some of the system design parameters (a_i) used in this cost modeling are:

1. Mission monetary cost - Includes launcher cost, spacecraft bus, mission operations and communications subsystem payload;

2. Total ΔV (change in velocity, which is directly related to fuel cost) budget required over the lifetime of the spacecraft - Includes evaluating minimum launch energy given specified epoch launch and arrival dates, Mars orbit insertion ΔV, and station keeping ΔV;

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[1]Tai, W.K., Mars Communication Network Design Trade Study, Masters Thesis, San Jose State University Department of Aerospace and Mechanical Engineering (In progress)

3. Communications link capability - The highest data rates that can be passed along the link with a given error rate. This capability is dependent upon satellite parameters such as antenna size, transmitter power, and distance to the receiver. These parameters drive such items as solar panel size, spacecraft mass, and launch vehicle selection. A link analysis was performed;

4. Communications satellite visibility - Capability from a specified site to view at least one satellite in the constellation at or above a specified elevation. It is another item which is affected by many parameters. This trade study examines several constellation configurations, using a common method to determine the amount of time a satellite is in view of a point on the surface. Ephemeris is calculated for each satellite of a constellation (up to six satellites). Coverage is calculated using a program designed to accept ephemeris input from several sources, and was analyzed for zero, ten, and thirty degrees above the horizon;

5. Spacecraft lifetime - Dependent on the spacecraft orbit and ΔV requirements;

6. Fault tolerance and reliability - The ability of a satellite network to perform its mission objectives after a failure of one or more satellites, or parts of a satellite, during the lifetime of the communication network;

7. Graceful degradation - The ability of a satellite network to perform its mission objectives as its performance degrades as the satellite ages and parts fail during the spacecraft lifetime;

8. Spacecraft design advantages and disadvantages;

9. Time required to establish and to construct network.

The weighting factors (A_i) are designed to give the customer flexibility in determining their "correct" answer, based upon their priorities. One customer may view money as no object, with their top priority being the time taken to establish the network. Another customer may be cash-limited, willing to wait the time required in order to afford the network. By including these weighting factors, the equation becomes customizable for a particular customer's requirements, and can provide a "least cost" solution appropriate to that customers constraints. The continuation work[2] will present several weighting factor sets, representing different user priorities (e.g. money is driving factor, coverage next, or time most important, cost next, etc.). Each set of weighting factors is used to generate a final score, and a 'best' configuration for that set of user priorities is presented. Several such user priority sets are presented, with their 'best' answer.

Previous Contributions

There have been several studies on topics related to this thesis. Of great value were the multitude of papers by JPL authors on the MESUR mission, which were used as

[2]Tai

a "sanity check" while calculating the many communications link results in this paper[3,4,5].

The inspiration for this work was contained in the paper by Pernicka, Henry, and Chan[6], which discussed the use of a Libration Point orbiter as a communications relay satellite. The idea expressed in that paper has been greatly expanded and studied in detail. Another paper with a large impact on this work was that by Draim[7], detailing a constellation configuration which would provide global communications coverage using four satellites. A modified version of this constellation configuration was one which was examined during the course of this study.

An interesting alternate approach was provided by Svitek, et al.[8], who proposed a very small, very limited, though very low cost, Mars Relay Satellite. This configuration was not examined closely in this paper, though will be in the continuation of this work[9]

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[3]Noreen, Gary K, MESUR Network Strawman DTE Telecom Design, JPL Interoffice Memorandum 3392-93-73, Jul 21 1993

[4]Noreen, Gary K, Mars Relay Satellite Configurations, JPL Interoffice Memorandum 3391-93-84, Nov 2 1993

[5]Martin, W. and Kantak, A., Analysis of MarsNet Lander - Relay Telecommunications Link, JPL Interoffice Memorandum 077IOM92.WLM, Oct 12, 1992

[6]Pernicka, H., Henry, D., and Chan, M., Use of Halo Orbits to Provide a Communication Link Between Earth and Mars, AIAA 92-4585-CP, AIAA/AAS Astrodynamics Conference, 1992

[7]Draim, J.E., A Common-Period Four-Satellite Continuous Global Coverage Constellation, *Journal of Guidance, Control, and Dynamics*, Vol. 10, No. 5, Sept-Oct 1987, 492-499

[8]Svitek, T. et al, Mars Relay Spacecraft: A Low Cost Approach, Ninth Annual AIAA/Utah State University Conference on Small Satellites, Logan, Utah, Sept 18-21, 1995.

[9]Tai

Organization

This thesis is organized as follows:

Chapter 2 describes the configurations of the various constellation configurations.

Chapter 3 details the constellation availability analysis, the study of which constellation provides the most opportunities for a ground user on Mars to make use of the constellation.

Chapter 4 provides an analysis of the communication links which would be used by the ground user to the relay satellite, and the relay satellite to Earth.

Chapter 5 summarizes the rest of the work on this project, accomplished by other team members.

The appendices provide amplifying information on several portions of this work. These include:

Appendix A provides more detail on the simplifying assumptions used to derive the Circular Restricted Three Body Problem equations of motion for the integration of Halo orbits.

Appendix B describes the CVG program, written by the author to perform communications coverage analysis on both Libration Point orbits (Halo) and Mars-centered orbits.

Appendix C provides a glossary of terms used in this work.

Appendix D details the modifications made to the constellation, described by Draim in his paper[10], to use his Earth-based constellation design at Mars.

Appendix E summarizes the spacecraft configurations used for each scenario, the advantages of that scenario, and its disadvantages. A short discussion of fault tolerance and graceful degradation are also included.

[10]Draim

CHAPTER 2

Constellation Configurations

In this study, the authors have examined several configurations for the Mars-orbiting portion of the system. These configurations have been "scored" against each other to determine which is the "best" for a given set of customer priorities. They all have benefits in some areas, and drawbacks in others. These configurations are described below.

Constellation Descriptions

Multiple, Low Orbiting Satellites

This configuration consists of several (four to six) communication satellites in low Mars orbit. Low in this case is from an altitude of approximately 300 to 1000 km. These satellites are phased evenly in their orbits, so that they do not all arrive at the polar regions at the same time. From this low altitude, the satellite can see a portion of the surface of the planet, several hundred kilometers in diameter. Their orbital period will be on the order of 90 to 120 minutes, approximately 9 to 13 revolutions per Sol (Mars day).

Orbital Configuration. The semi-major axis of the orbit was chosen to provide an integral number of revolutions in one Sol. By using the relationship relating orbital period to semi-major axis, $a = \left(\frac{P * \sqrt{\mu}}{2\pi} \right)^{2/3}$ it became a matter of selecting a rough altitude, looking at

the table, and determining the appropriate semi-major axis for the orbit (see Table 2-1). Orbits with an integral number of revs per Sol were chosen to give the orbital coverage repeatability.

Table 2-1 - Number of Orbits vs Satellite Altitude (km)

Number of Orbits per Sol	Period (sec)	Semi-major Axis (km)	Altitude (km)
1	88642.00	20462.76	17069.76
2	44321.00	12890.73	9497.73
3	29547.33	9837.47	6444.47
4	22160.50	8120.65	4727.65
5	17728.40	6998.16	3605.16
6	14773.67	6197.22	2804.22
7	12663.14	5591.98	2198.98
8	11080.25	5115.69	1722.69
9	9849.11	4729.36	1336.36
10	8864.20	4408.57	1015.57
11	8058.36	4137.16	744.16
12	7386.83	3904.00	511.00
13	6818.62	3701.14	308.14

Table 2-1 - Number of Orbits vs Satellite Altitude (km)

Number of Orbits per Sol	Period (sec)	Semi-major Axis (km)	Altitude (km)
14	6331.57	3522.73	129.73
15	5909.47	3364.37	-28.63

As can be observed, beyond a certain point it is impossible to get any additional revolutions without impacting the planet. Beyond thirteen revs per day, the orbit gets too low to be of practical use. For this particular case, two configurations were chosen for analysis; ten and thirteen revs per day. These use a semi-major axis of 4408.57 km and 3701.14 km, respectively. Two altitudes were analyzed to see what variation there would be between a "low" 308 km altitude and a "medium" 1015 km altitude.

Two separate configurations were examined for each altitude: a constellation of four satellites in two orbital planes; and a constellation of six satellites in three orbital planes. The orbital planes are inclined 90°, which provides coverage of Mars' polar regions. Each orbital plane contains two satellites, 180° apart. This provides for some backup coverage in case of a satellite failure (this will be discussed in greater detail in Appendix E, Satellite Configurations for each Scenario). Each orbital plane is phased with respect to the others, by offsetting the longitude of the ascending node (0°, 60°, and 120° for three planes, or 0° and 90° for two planes).

Common-Period Inclined Orbits

The Common-Period, Inclined configuration (CPI) was proposed by John Draim in his 1987 paper[11]. His paper defined a "Draim" constellation, which would provide continuous coverage of the Earth using only four satellites. The satellites are in elliptical orbits (e = 0.263), inclined 31.3° to the equator. Draim computed the required semi-major axis to be a minimum of 24316 n mi, based on a trial and error geometric simulation. This was adequate for the Earth, but this work required a Mars-centered constellation. Since Draim's method of computing the semi-major axis was not easily repeatable, and was determined geometrically, the author made an approximation of the proper orbit by scaling the Draim Earth constellation by a factor equal to the ratio of Mars' radius to Earth's radius (3393/6378 = 0.531). The inclination and phasing were not changed from Draim's paper. This constellation was run through a coverage simulation (the simulation program, CVG, is described in detail in Appendix B), and the result examined for suitability. After several corrections, a set of orbits was found around Mars which approximated Draim's results for Earth. The final orbits had a slightly larger semi-major axis than the scaled value initially calculated, but were the same in all other orbital characteristics.

Aerosynchronous Orbit

The aerosynchronous case is simply the extension of Earth's geosynchronous orbits to Mars. The satellites are placed at an orbital radius of approximately 20,462 km

[11]Draim.

(see Table 2-1, above, one orbit per Sol), at which their angular velocity matches that of the planet's rotation rate. As in the above case of multiple, low orbiting satellites, two configurations were analyzed: a three satellite constellation and a four satellite constellation. The satellites are spaced equally around the orbit, either 120° or 90° apart. These satellites have the advantage of always being in view of a particular area of the planet. This simplifies the pointing requirement of a ground station, because the satellite is "always" in the same place in the sky. It does so at the cost of polar coverage, which this constellation provides little of.

Halo Orbit

The halo orbit is one of the more interesting of this study. Rather than orbit Mars itself, the two satellites in this constellation orbit two points in space known as Lagrange Points (or L points). Lagrange points are discussed in detail in Chapter 3. The Lagrange points are equilibrium points in "Three-body" orbital systems. These three bodies consist of two "primaries" (a smaller Secondary and a larger Primary) and a much smaller "satellite," or third body.

The smaller primary orbits the larger primary body. In this work, the primaries are the Sun and the Mars. Mars' orbit about the Sun can be described using two-body equations, if the satellite is sufficiently small that its mass does not perceptibly affect the motion of the two larger bodies (as in the case of an Earth-launched satellite, compared to Mars and the Sun).

By placing the satellite at just the right point in space, moving with just the right velocity, a halo orbit can be achieved. The halo orbit, a special case of the Lissajous orbit, is an orbit which circles the Lagrange point, and the Sun-Mars line on which it lies. The halo orbit is shaped like a hoop circling the Lagrange point. The advantages of such an orbit are 1) that a satellite there can see almost half of the planet's surface, 2) the satellite is always near the Sun-Mars line (which simplifies tracking the satellite, though increases the background noise the satellite's signal must overcome), 3) the resultant net force on a satellite there is very small, so the satellite's station keeping budget is much smaller, and 4) power generation is simplified.

Phobos - Deimos Landers

The final constellation "configuration" examined in this work is placing a lander on the Martian moons Phobos, Deimos, or both. A lander placed on one of the moons would orbit the planet .8 (Deimos) or three times daily (Phobos). As the moons orbit at a low inclination, these satellites provide little if any polar coverage, much the same as aerosynchronous satellites.

CHAPTER 3

Constellation Availability Analysis

Problem Description

Constellation Availability is defined as the percentage of time at least one satellite in the constellation is visible to a "user" on the ground. Each constellation type provides a different level of service to the user (different configurations within the same constellation type can also provide differing levels of service). This parameter is used to help measure the ability of a particular constellation type/configuration to provide service to the user on the ground.

Terms Defined

The following are some terms used in this chapter and their definitions:

Satellite Vector: the vector from the center of the planet to the satellite.

Station Vector: the vector from the center of the planet to the user.

Station-Satellite Vector: the vector from the ground observer to the satellite, equal to the vector difference between the Satellite Vector and the Station Vector.

Constellation: the group of communications satellites.

Constellation Availability: Constellation Availability is the percent time that at least one satellite in the constellation is visible to the "user." It is determined by calculating the elevation of each satellite at multiple user locations, and tallying which would have a view of any one of the satellites in the constellation. The satellites are then advanced in their orbit over a small amount of time (called a timestep), and the process is repeated. At the end of the desired timespan (number of timesteps), the tally for each ground point (how many times a satellite was visible there) is divided by the total number of timesteps (the number of opportunities for visibility). The result is the percentage of time that a user at that point had access to the constellation.

Satellite Vector Generation

The coverage program described in this chapter (referred to as "CVG" for the remainder of the chapter) requires, as an input, a position vector for each satellite in the constellation being simulated. This vector can be specified in cartesian coordinates, as in the case of the Mars-orbiting satellites, or coordinates in a rotating reference frame, as in the case of the halo, or Lagrange point orbiters.

Cartesian Vector Generation. The Mars-orbiting satellites use a cartesian state vector as input to CVG. The X, Y, and Z positions are the only portion of the state vector that are actually used, as they alone determine where the satellite is in relation to the planet center. The vectors are generated using POHOP, a program from the Jet Propulsion Laboratory, Pasadena, CA. By specifying the initial orbit and starting position (Keplerian coordinates), and the time interval, POHOP will propagate the orbit, generating an output file containing time, position, and velocity information (POHOP can be configured to output several other parameters, as well. They were not required for this application, and were not requested of POHOP).

Rotating Frame Vector Generation. The rotating reference frame is introduced to allow the simplification of the three-body problem and facilitate finding solutions. First, a review of the three-body problem is needed to fully understand the halo orbits described later in this work.

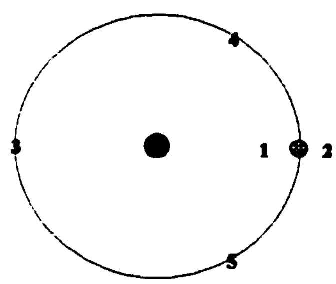

Figure 1 Illustration of the positions of the Lagrange points L_1 through L_5. The large circle in the center represents the central body of the two body system. The smaller circle represents the smaller of the two bodies, with the circle it lies on representing the orbital path of the second body. The numbers represent the approximate positions of the corresponding L-point (1 for L_1, 2 for L_2, etc.).

Three-body problem. When two bodies orbit each other, their motion can be described using Kepler's laws. Issac Newton solved for the motion analytically, giving rise to the Two Body Problem and its solution. When another body is added, the Three Body Problem is defined.

As stated in the introduction of the paper, there is no known general closed-form solution to the three-body problem. Many astrodynamicists have spent years of time devoted to just such a search. By simplifying the problem somewhat, several particular solutions to the problem can be determined. By restricting the mass of the third body to being small enough that its presence will not perturb the motion of the two "Primaries" ($M_3 << M_2 < M_1$), and limiting the Primaries to circular orbits, the problem becomes known as the Circular Restricted Three Body Problem (CR3BP). A mathematician by the name of Lagrange determined that the CR3BP does have five "equilibrium" points, called "Lagrange Points" or L-points.

L_1, L_2, and L_3 all lie on the line connecting the two Primaries. L_1 lies between the two bodies, closer to M_2. L_2 lies nearly the same distance on the opposite side of M_2 (outside the orbit of M_2). L_3 lies opposite M_2, on the far side of M_1. L_4 and L_5 each lie a distance equal to the distance between M_1 and M_2 ($R_{M1\text{-}M2}$) from M_1, 60° ahead of (L_4) and behind (L_5) M_2. At each of these points, the gravitational and centrifugal forces acting on M_3 balance, and the body will remain at this position. These are unstable equilibrium points, though, and much like a marble at the top of a hill, the slightest perturbation will send the body flying away. Fortunately, moderate control will maintain M_3 at the libration point. In this application, we will only be considering the L_1 and L_2 points.

<u>Coordinate System.</u> The coordinate system used in this problem is referred to as the Rotating Reference Frame. The origin is at the barycenter (mass center) of the M_1-M_2 system. The X-axis is defined in the direction of M_2, and rotates as M_2 orbits M_1. The Z axis is defined using the right hand rule (out of the page with M_2 orbiting M_1 in the

counter-clockwise direction). The Y axis is defined to make a right-hand coordinate system. Since the Lagrange points move around the Sun as M_2 orbits M_1, by using a rotating coordinate system, we can show the L point locations as fixed with respect to the coordinate system (since the orbit is circular, R_{M1-M2} is constant, and the positions of the L points do not vary). Now, the only body in motion in the system is M_3.

The scale of distance and time are different in the rotating frame. It is a canonical coordinate system, i.e. time, distance, and any quantities derived from these two (velocity, acceleration, etc.) become unitless. The distance unit is defined as R_{M1-M2} (so M_2 is at $(1.0 - \mu)$ distance units along the X-axis from the origin). One time unit equals the time it takes for M_2 to move π radians in its orbit around M_1. By using such a system, the identities of the bodies can be changed merely by varying a parameter μ, the mass fraction of M_2 ($\mu = M_2/(M_1+M_2)$). The positions and velocities of M_3 are output in this coordinate system.

Integrating the Orbit. Once the positions of L_1 and L_2 are determined, orbits about each can be found. Analytic approximations can be used to relate the size of the X, Y, and Z dimensions of the halo orbits. Using some mission parameter to determine the size of one dimension, the appropriate sizes of the other dimensions can be easily determined. These equations require proportionality constants, which can be estimated analytically. If others have studied the particular system before (i.e. Earth-Moon, Sun-Earth, etc.), the proportionality constants might have already been determined by numeric means. If not, the only way to find the orbit is trial and error (i.e. enter starting conditions, integrate the orbit, see if it is periodic, and modify the initial conditions if necessary).

It is possible to use a method called differential corrections to aid in determining proper initial conditions for a periodic orbit. This method involves integrating an additional 36 equations (along with the initial six simultaneous equations) through half the orbit, and comparing the final conditions with known midpoint conditions (M_3 should only be moving in the Y direction as it crosses the X axis half way around the orbit). By using the difference between the expected and actual X and Z velocities, and multiplying this two-element vector by a correction matrix extracted from the additional 36 integrated equations, corrections to the initial conditions are calculated. These are applied, and the process started again. By automating the process in the integration program (stopping the iterations when the error drops below a tolerance), much of repetitious work is eliminated, and the process is sped up immensely. It must be mentioned that the differential corrections process might not converge (the error might never drop below the tolerance). This was the result in the author's case. The only option, then, is to manually try different initial conditions and hope to close to a usable orbit. This can be a time consuming process, but one which, in this case, resulted in success.

Discussion of CVG Program

There are a great many commercial tools available that will calculate availability for a given constellation. There are two reasons why a custom program was written: first, the cost of these programs (several hundred to thousands of dollars) was prohibitive; and second while these type programs are undoubtedly quite good at calculating constellation parameters for satellites orbiting another body (e.g. the Sun, Earth, or Mars) , the author did not believe that they would react well to a satellite in a Lagrange orbit. Once the decision was made to use a custom program on the halo case, the program was configured to accept either orbital case (halo or Mars-orbiting) as inputs, so that the results could be directly compared. For a more detailed discussion of the program, please refer to Appendix B.

Program Inputs. CVG was designed to accept as input a constellation of up to six satellites, and an indicator letting the program know if the satellites are in a Mars-centered orbit or a halo (Lagrange) orbit. The program needs to know the type of orbit because, if a halo orbit, there are coordinate transformations which must be applied to the input vector before it can be used by the program (POHOP outputs data in a Mars-centered inertial reference frame, and the halo integrator outputs data in the rotating frame described above ["Coordinate System"]). There are other differences in how the program handles the two types of orbits. Because the spacecraft in a halo orbit moves so slowly (it makes only two complete orbits for every orbit M_2 makes of M_1), CVG only updates the position of M_3 every "day" (88642 seconds in this case). When considering a Mars-centered constellation, CVG updates the position of the satellite every 10 minutes (the distance moved during that time is much more significant because the satellites are moving much more quickly in Mars-centered orbits). This value was chosen as a compromise between simulation fidelity and run-time.

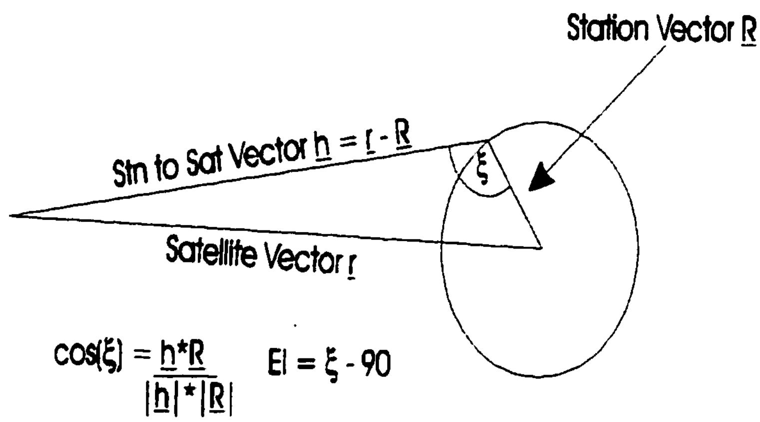

Figure 2 Vector diagram of the elevation angle calculation process.

<u>Program Algorithm.</u> Once the program knows what type of orbit has been input, it handles all constellations in roughly the same manner. New satellite positions are read in (up to six satellites). Coordinate transformations are performed if required. The program then begins the process of determining whether any of the satellites can be seen from points on the "surface" (a sphere of approximately 3393 km radius).

The program begins at the North Pole of Mars. A grid of latitude and longitude is established, one degree by one degree (180° x 360°). CVG begins moving along the grid, looping through 360° longitude for a specific latitude, incrementing the latitude, and beginning again. At each point, the program determines the Station vector (see "Terms

Defined"), then the vectors from the Station to each satellite. By taking the vector dot product of the Station vector and the Station-to-satellite vector, the angle between the two can be determined. With that angle known, determining the elevation of the satellite above the "horizon" is a matter of subtracting 90° from that result. The program then compares the elevation to the maximum value seen during this set of calculations (the "max-el" is reset to zero for each new Station position), and, if the new value is larger, stores it. Once it has examined all of the satellites in the constellation, the program has the elevation of the satellite highest above the "horizon" at that point on the surface.

Three "counter" matrices (360 x 181, for each latitude/longitude point, to include the equator) are established at program initialization to store visibility information. The first is used to store whether at least one satellite in the constellation was visible at greater than zero degrees (max-el > 0°) at that point on the ground, the second for ten degrees or higher (max-el > 10°), and the third for greater than thirty degrees (max-el > 30°) {thirty degrees is the elevation mask used in several NASA and JPL papers reviewed in the course of this work}. If the maximum elevation meets these criteria, the information is stored, and the program moves to the next position. Once the planet has been circled, the process begins again, and continues until there is no longer any data remaining in the first input file. As a note, the program was optimized so that, if the latitude were North or South 90° (the poles) it does not run through each longitude point, as the Station vector will be the same in all 360 cases. It merely makes the calculations once. This way, 718 sets of calculations (up to 18 equations per set) are saved for each ten minute increment (called a timestep), or up to 1,912,752 calculations per simulated day. These savings are significant when analyzing a halo constellation, which is run for approximately 687 simulated days (saving 438,020,208 calculations). A more detailed description of the algorithm is contained in Appendix B.

Discussion of Scoring Criteria

Each constellation configuration is scored in several areas. Those areas are: Polar Region Coverage, Temperate Region Coverage, and Equatorial Region Coverage. These regions were selected because:

a) they are relatively easy to calculate,

b) they are intuitive to the reader,

c) they follow from users requirements and targeting specifications.

The Polar Region extends from the North Pole to 67° North latitude, and from 67° South latitude to the South Pole. These demarcations were chosen based upon Earth's Polar Circles.

The Temperate Region is equivalent to the area between Earth's Polar Circles and the Tropics, extending from 67° latitude to 23° latitude, North and South.

The Equatorial Region covers all the area not previously mentioned, from 23° North latitude to 23° South latitude, and is equivalent to the Earth's Equatorial region.

Discussion of Results

The results of the CVG simulations are presented below. The simulation for each constellation configuration resulted in three matrices, each containing 181 x 360 floating point numbers between 0.0 and 1.0, representing the percentage of time that a satellite in the constellation was visible at that point on the "surface" of Mars. Two of the matrix elements (specifically two elements of the 360 which represent each pole, at which point the data is redundant) were replaced with the values 0.0 and 1.0. This is to ensure that the full range of desired values was contained in each file, so that the results are displayed in a consistent manner.

The data are read into MathCAD 4.0®, by MathSoft. They are then displayed using the color contour chart function of MathCAD®, copied and pasted into Corel Draw 6.0®, where the axis labels are added, and then imported into this document. The color contour chart function reads the data, and assigns colors based on the highest and lowest data points seen. For this reason, the above mentioned values of 0.0 and 1.0 were added to each matrix, to ensure that each scenario had the same high and low point.

Each constellation is described in summary below. Its contour maps are displayed, along with the constellation's scores. Figure 3 shows a legend for the following maps, with blue/purple representing 0%, and red representing 100%.

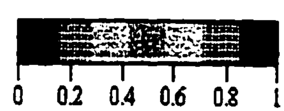

Figure 3 Legend. Colors represent percentages on coverage maps.

Halo Constellation. The satellites in the Halo constellation are located approximately 1 million kilometers from Mars. At this distance, they are visible to nearly half the surface of the planet. With one relay positioned on the day side of Mars, and the other on the night side, the pair enjoy a nearly total view of the planet. There is a small "band" between the Fields of View (FOV) of the two satellites which is not visible to either satellite. Because this band runs nearly perpendicular to the equator, no area is greatly affected by it, as points on the surface rotate through the band as the planet turns on its axis. This small "gap" in coverage lasts approximately 100 seconds, from one satellite disappearing below the horizon until the other rises above the other horizon, or approximately 4 hrs from the time one satellite drops below 30 degrees in elevation until the other rises above 30 degrees (8 Hrs per day total lack of visibility above 30 degrees). These larger "gaps" in coverage will only affect landers situated near large obstructions, such as cliffs or mountain ranges, and then only those very close to them (for instance, to be obstructed to 30 degrees elevation, a lander would have to be closer than 2000 m to the high point of an obstruction which rises 1 km above the level of the lander). Typically, mission planners would not choose such a site for a lander, preferring a flat plain, and targeting areas closer to the center of the plain. Earth-based ground station sites are also chosen carefully, and typically see no more than five degrees of obstruction (which would result in approximately 45 minutes of obstruction twice per Sol).

Figure 4 shows the coverage results for the halo orbit at zero degrees elevation or above. The constellation provides coverage values as follows:

Polar Zone - 99.1%;

Temperate Zone - 99.6%;

Equatorial Zone - 99.7%.

This map, as stated before, does not take into account any blockage by hills, rocks, etc. Figure 5 shows data for coverage at 30 degrees elevation or higher,

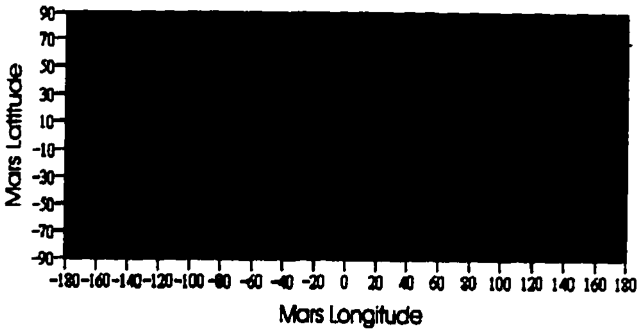

Figure 4 Constellation Availability results for Halo Constellation at zero degrees elevation of better. Polar Region 99.1%; Temperate Region 99.6%; Equatorial Region 99.7%.

which will take into account such obstructions. The coverage values for this elevation are:

Polar Zone - 18.4%;

Temperate Zone - 42.3%;

Equatorial Zone - 59.5%.

The numbers in Figure 5 are significantly lower than those in Figure 4. Because the satellites must get higher in the sky before

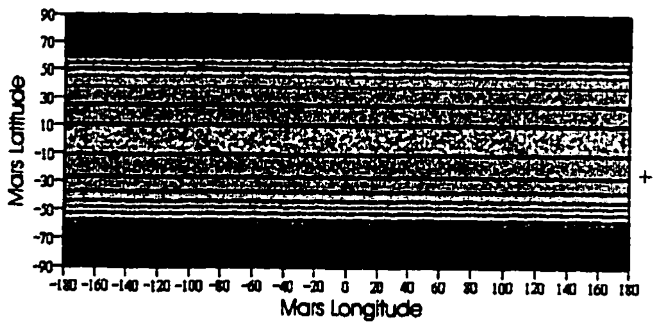

Figure 5 Constellation Availability results for Halo Constellation at thirty degrees elevation or better. Polar Region 18.4%; Temperate Region 42.3%; Equatorial Region 59.5%.

they are counted in Figure 5, they spend more time below the desired elevation, and reduce the amount of time they are above it.

Also illustrated by Figure 5 is the "banding" of coverage by the Halo Constellation. This is seen because the satellites are moving relatively slowly in space, compared to the planet's rotation beneath it. Contrast this with Figures 6, 7, 8, and 9 (the

Aerosynchronous Constellations), where the satellite rotates with the ground point, and gives "patches" of coverage.

Aerosynchronous Constellation. The Aerosynchronous constellation is equivalent to those which orbit the Earth along its equator ("Geosynchronous"). They orbit once per Sol, at a speed such that they remain over one region of the planet's surface. This is a preferred configuration for a communications satellite, because ground user pointing is greatly simplified. This work examined two aerosynchronous configurations, a three-satellite constellation and a four satellite constellation.

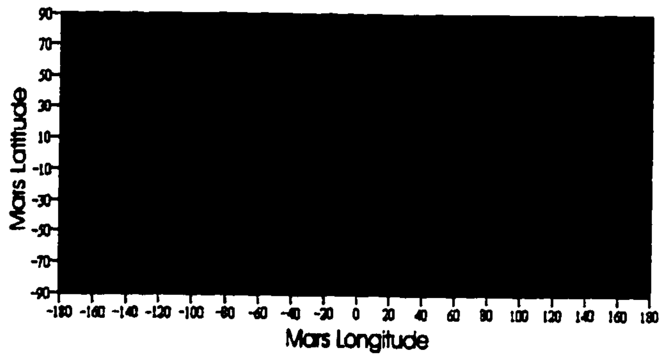

Figure 6 Constellation Availability results for three-satellite Aerosynchronous Constellation at zero degrees elevation or better. Polar Region 49.9%; Temperate Region 100%; Equatorial Region 100%.

Three Satellites. The coverage results from the three satellite constellation are summarized below. This, and the four satellite case, were the testbed cases. As synchronous satellites are used here on Earth, their parameters, and the expected coverage distributions, are well known and served to troubleshoot and validate the CVG program. Figure 6 illustrates the results calculated for the three-satellite case. They are:

Polar Zone - 49.9%;

Temperate Zone - 100%;

Equatorial Zone - 100%.

As the figure shows, equatorial coverage is very good, at the expense of polar visibility. Because the satellites never rise far above the equator, they never have a good view of the polar region. Although the "Polar Zone" number was 49.9%, the constellation achieves this by providing almost total coverage below approximately 81 degrees latitude, and no coverage above that point.

At thirty degrees, the numbers are, as expected, lower. Figure 7 illustrates the results of this simulation.

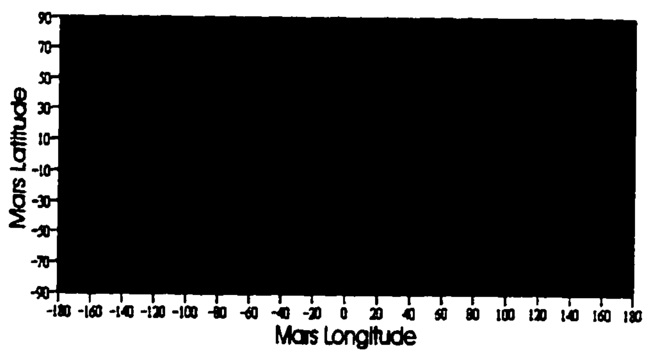

Figure 7 Constellation Availability results for three-satellite Aerosynchronous Constellation at thirty degrees elevation or better. Polar Region 0.0%; Temperate Region 39.8%; Equatorial Region 84.3%.

Polar Zone - 0.0%;

Temperate Zone - 39.8%;

Equatorial Zone - 84.3%.

As the figure shows, the regions of good coverage have been greatly reduced. Polar coverage is non-existent above approximately 50 degrees latitude, and pockets along the equator have no coverage, either. This does not mean that users here would not "see" the satellite, but that the elevation of a satellite in their area is not above thirty degrees. This would tell a user to expect that, if a lander were to land near some obstruction, it might not be able to view a satellite. As mentioned above, that likelihood is small.

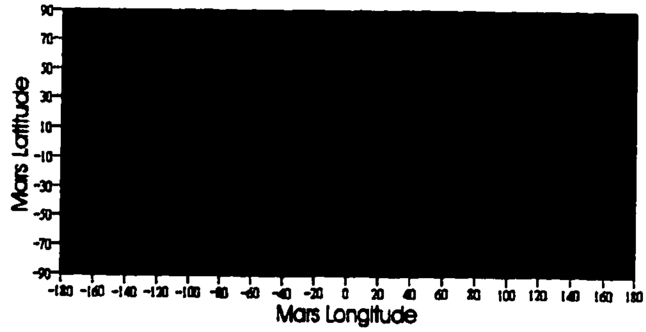

Figure 8 Constellation Availability results for four-satellite Aerosynchronous Constellation at zero degrees elevation or better. Polar Region 55.7%; Temperate Region 100%; Equatorial Region 100%.

Four Satellites. The four-satellite case adds one more satellite to the constellation, and phases the satellites for a more balanced operation. A large advantage of this type of constellation is that all of the satellites are in the same orbital plane. This makes it possible, in the event of a satellite failure or anomaly, to rephase the constellation and move to a three-satellite operation, with coverage results shown above.

Figure 8 illustrates the results of the zero-degree elevation simulation.

Polar Zone - 55.7%;

Temperate Zone - 100%;

Equatorial Zone - 100%.

The increased numbers in the Polar Zone are due to filling the gaps visible in Figure 6. By phasing the satellites closer together, the gaps were closed, and Polar coverage improved. This constellation, like the three satellite constellation, does not provide coverage to ground sites above approximately 81 degrees latitude, which would rule out its use to support polar landers.

Figure 9 shows the improved coverage provided above thirty degrees in elevation. Its results are as follows:

Polar Zone - 0.0%;

Temperate Zone - 52.6%;

Equatorial Zone - 100%.

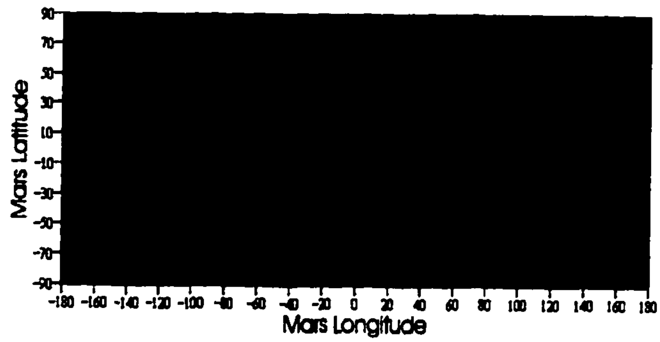

Figure 9 Constellation Availability results for four-satellite Aerosynchronous Constellation at thirty degrees elevation or better. Polar Region 0.0%; Temperate Region 52.6%; Equatorial Region 100%.

Again, this configuration provides improved equatorial coverage, compared to the three-satellite case. It is still lacking, though improved, in the temperate and polar regions. Because of the nature of synchronous satellites, these zones will remain relatively stationary. This would lead mission planners to target their landers to covered areas to ensure communications, rather than to let the mission goals determine the targeting and bringing the communications in to serve the customer. This would not seem to be the best way to plan the exploration of a new planet.

<u>Common Period, Inclined Constellation.</u> The Common Period, Inclined constellation, as mentioned in Chapter 3, was proposed by Navy Captain John Draim. His proposal was simple: four satellites in inclined orbits, two with apogee over the northern hemisphere, two over the southern. By phasing these satellites properly, continuous coverage of the Earth's surface could be achieved. In this work, his orbits were modified to fit Mars (for

more details, please see Appendix D), and run through CVG. The results are shown in Figure 10:

Polar Zone - 100%;

Temperate Zone - 100%;

Equatorial Zone - 100%.

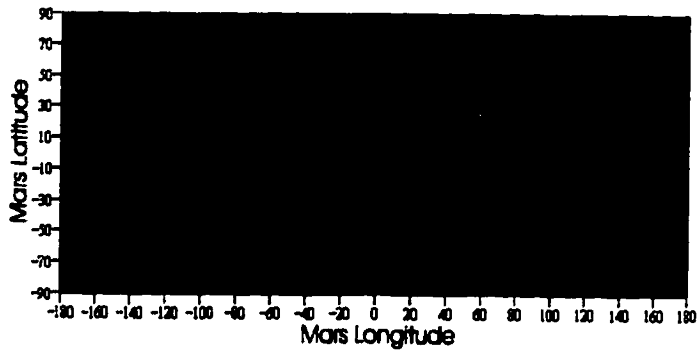

Figure 10 Constellation Availability results for four-satellite Common Period, Inclined Constellation at zero degrees elevation or better. Polar Region 100%; Temperate Region 100%; Equatorial Region 100%.

Complete coverage is achieved, in this case, because one of the four satellites can cover the gaps left between the coverage zones of the others. Please note: the 100% number does not allow for any obstructions in the Line-of-Sight, such as hills, gullies, etc.

Figure 11 shows the simulation's results above 30 degrees elevation:

Polar Zone - 29.1%;

Temperate Zone - 65.3%;

Equatorial Zone - 77.8%.

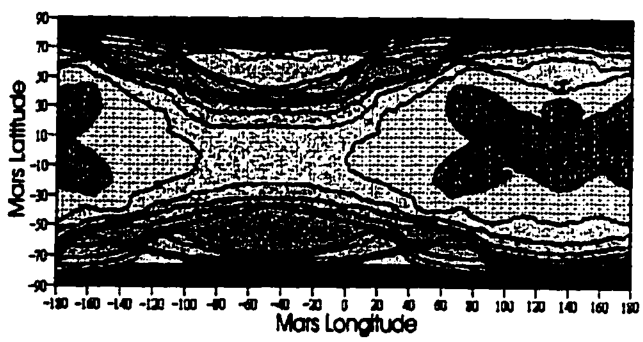

Figure 11 Constellation Availability results for four-satellite Common Period, Inclined Constellation at thirty degrees elevation or better. Polar Region 29.1%; Temperate Region 65.3%; Equatorial Region 77.8%.

The results show very high availability along the ground trace of the satellites as they approach apoapsis (where the satellite spends a lot of time due to its slow speed), and less availability as the satellites approach periapsis, as one would expect, due to the faster motion of the satellites (they spend less time near periapsis due to the greater speed at which they are moving).

By comparing these results to those in Table 3-1, it is observed that the CPI Constellation results are better than those of the Halo Constellation. It must be pointed out, though, that these results were obtained using four satellites, where the Halo Constellation used only two satellites.

Table 3-1 - Comparison of Halo and Common Period Inclined (CPI) Results

Zone	Halo Availability (% time)	CPI Availability (% time)
Polar (0°)	99.1%	100%
Temperate (0°)	99.6%	100%
Equatorial (0°)	99.7%	100%
Polar (30°)	18.4%	29.1%
Temperate (30°)	42.3%	65.3%
Equatorial (30°)	59.5%	77.8%

Low Orbit Constellation. Two Low Orbit Constellation configurations were examined in this work: a four satellite configuration, and a six satellite configuration. The two configurations were chosen to examine the increase in coverage gained by adding more satellites to the constellation. As illustrated in Chapter 3, the periods of these satellites were chosen such that an integral number of orbits could be completed per Sol. This allowed the simulation to be run for a shorter amount of time, just until the beginning of the next cycle (or one Sol, in this case).

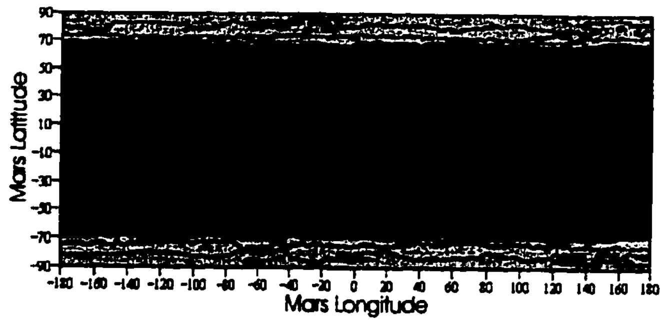

Figure 12 Constellation Availability results for four-satellite Low Orbit Constellation at zero degrees elevation or better. Polar Region 47.0%; Temperate Region 17.5%; Equatorial Region 11.5%.

Four Satellite. This configuration is made up of two orbital planes, each containing two satellites. The planes are phased 90° apart. Each satellite is phased 180° from its partner in the same plane, and 90° from its counterparts in the adjacent plane. This way, only one satellite passes over the pole at a given time. The orbital planes are inclined 90° to the equator, in a true polar orbit. This was done to simplify the simulation. Many more runs would be required to determine the optimal solution to this problem.

As seen in Figure 12, polar coverage will be greater here than in many of the other configurations, as a satellite passes over the pole four times in one orbital period. Compare that to having a satellite passing overhead only four times in one Sol, and it becomes obvious why the Polar Availability increases dramatically.

The results for the four-satellite, low orbit configuration at zero degrees elevation are:

Polar Zone - 47.0%;

Temperate Zone - 17.5%;

Equatorial Zone - 11.5%.

Figure 13 presents the result of the four-satellite, low orbit constellation at thirty degrees elevation. Its results are as follows:

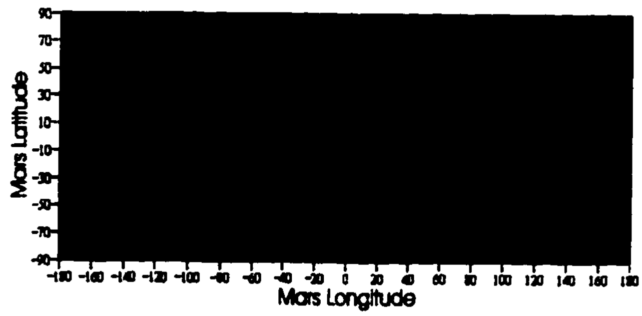

Figure 13 Constellation Availability results for four-satellite Low Orbit Constellation at thirty degrees elevation or better. Polar Region 7.8%; Temperate Region 1.7%; Equatorial Region 1.2%.

Polar Zone - 7.8%;

Temperate Zone - 1.7%;

Equatorial Zone - 1.2%.

These results are much lower than others previously presented because the satellites are so much lower in altitude than previous cases. They are primarily out of view, except for a total of perhaps 45 minutes per Sol. Of that, the satellites spend a great deal of the time below thirty degrees in elevation (climbing away from or falling back to the horizon). This is the primary disadvantage of this constellation type.

Six Satellite. The changes between this configuration and the previous were minor. Two satellites in an additional orbit plane were added. The three orbit planes were then spaced evenly around Mars. The satellites were phased so that they would not all arrive at the pole at the same time, and the simulations were run again. Figure 14 displays the results of the six-satellite constellation at zero degrees elevation.

Polar Zone - 67.6%;

Temperate Zone - 25.9%;

Equatorial Zone - 17.0%.

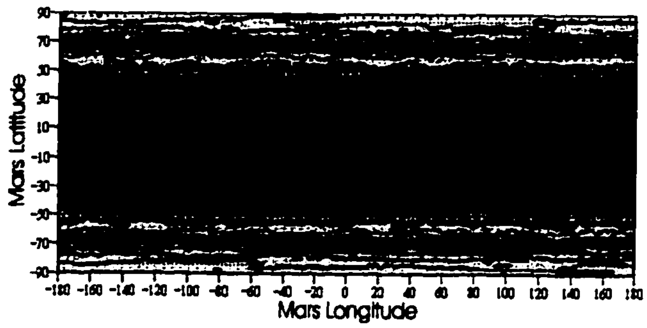

Figure 14 Constellation Availability results for six-satellite Low Orbit Constellation at zero degrees elevation or better. Polar Region 67.6%; Temperate Region 25.9%; Equatorial Region 17.0%.

The increase in performance over the four satellite configuration comes at the expense of two additional satellites and additional complexity in resource scheduling.

The results for thirty degrees elevation are similar to those for the four satellite configuration, with the expected increase in performance given by the additional satellites. Figure 15 illustrates the results.

Polar Zone - 11.7%;

Temperate Zone - 2.5%;

Equatorial Zone - 1.7%.

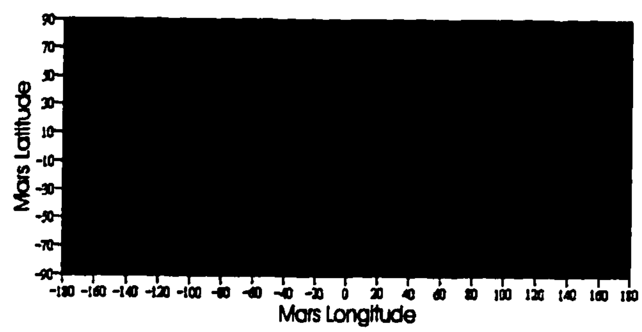

Figure 15 Constellation Availability results for six-satellite Low Orbit Constellation at thirty degrees elevation or better. Polar Region 11.7%; Temperate Region 2.5%; Equatorial Region 1.7%.

The low values for the 30 degree cases are, as mentioned above, caused by the amount of time the satellites spend rising high enough to be included in the count (in fact, on many of the "passes," the satellite will never rise above 30 degrees and be included at all).

<u>Medium Orbit Constellation.</u> The final constellation which was examined was the Medium Orbit. The configuration for this constellation is similar to that of the previous section, the Low Orbit. The main difference between the two is the altitude of the satellites, with the medium being approximately 700 km higher in altitude. This constellation type was also broken into two parts, the four satellite and the six satellite configurations.

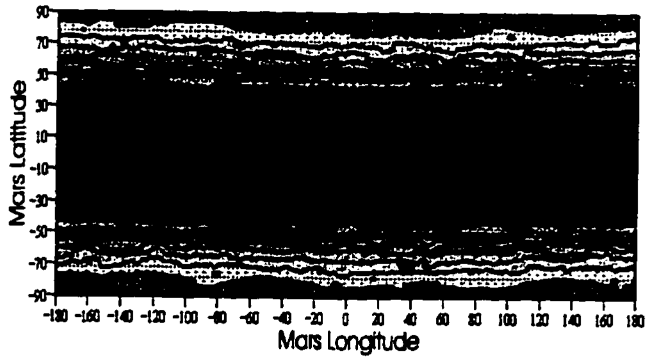

Figure 16 Constellation Availability for four-satellite Medium Orbit Constellation at zero degrees elevation or better. Polar Region 83.2%; Temperate Region 50.5%; Equatorial Region 32.5%.

Four Satellite. The results in this case are similar to those in Figure 12, the four-satellite low orbit constellation, with the difference that the higher altitude provides much longer visibility periods for each satellite, and, in the thirty degree case, each satellite is more likely to rise high enough to be included in the count. Figure 16 provides the results of this simulation.

Polar Zone - 83.2%;

Temperate Zone - 50.5%;

Equatorial Zone - 32.5%.

Figure 17 presents the simulation results for the thirty degree case.

Polar Zone - 32.7%;

Temperate Zone - 10.1%;

Equatorial Zone - 6.8%.

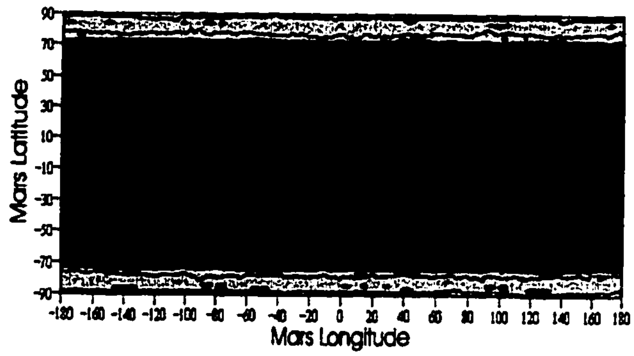

Figure 17 Constellation Availability results for four-satellite Medium Orbit Constellation at thirty degrees elevation or better. Polar Region 32.7%; Temperate Region 10.1%; Equatorial Region 6.8%.

Comparing the four satellite medium orbit constellation to the six satellite low orbit constellation, it is apparent that the increase in altitude more than compensates for the reduction in the number of satellites, providing many more opportunities for access to the constellation.

Six Satellites.

Examining the six satellite, medium orbit constellation, it is expected that the results will show an improvement over the four satellite case, which itself was an improvement over the low orbit cases, having the

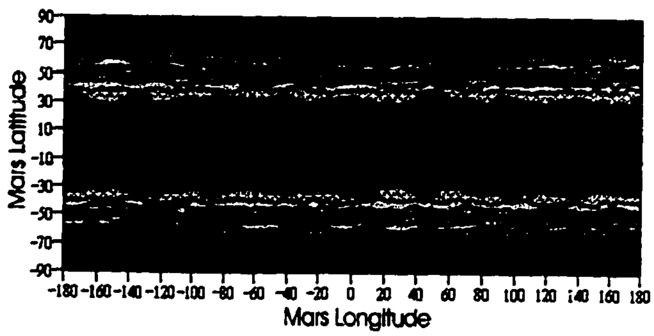

Figure 18 Constellation Availability results for six-satellite Medium Orbit Constellation at zero degrees elevation or better. Polar Region 98.6%; Temperate Region 68.0%; Equatorial Region 47.7%.

benefit of both higher altitude and more satellites. Figure 18 presents the zero degree elevation case.

Polar Zone - 98.6%;

Temperate Zone - 68.0%;

Equatorial Zone - 47.7%.

As the figure shows, this configuration provides near constant coverage over the poles, while sacrificing some coverage at the equator. Examining the thirty degree results, Figure 19 displays the results of the simulation run.

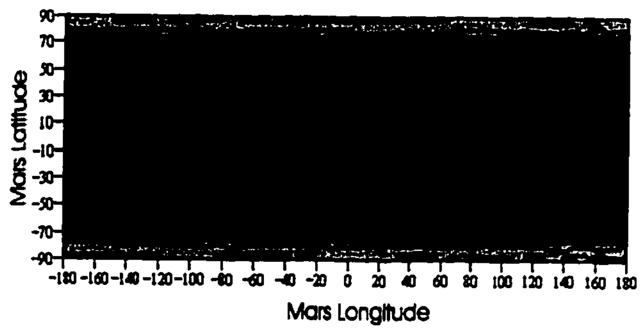

Figure 19 Constellation Availability results for six-satellite Medium Orbit Constellation at thirty degrees elevation or better. Polar Region 48.7%; Temperate Region 15.1%; Equatorial Region 10.0%.

Polar Zone - 48.7%;

Temperate Zone - 15.1%;

Equatorial Zone - 10.0%.

As shown in Table 3-2, the six satellite medium orbit configuration provides the best coverage of the low/medium orbit cases, providing a vast improvement over the four satellite case, but at the cost of increased time building the constellation and an increased cost for the constellation.

Conclusion. In concluding this chapter, Table 3-2 will directly compare the results from each section. As illustrated below, it appears that the Halo, the CPI, and the Medium 6 orbits are the leading candidates for coverage considerations.

Table 3-2 - Comparison of Constellation Availability Results (%time)

Zone	Halo	CPI	Aero	Low 4	Low 6	Med 4	Med 6
Polar (0°)	99.1%	100%	49.9%	47.0%	67.6%	83.2%	98.6%
Temperate (0°)	99.6%	100%	100.0%	17.5%	25.9%	50.5%	68.0%
Equatorial (0°)	99.7%	100%	100.0%	11.5%	17.0%	32.5%	47.7%
Polar (30°)	18.4%	29.1%	0.0%	7.8%	11.7%	32.7%	48.7%
Temperate (30°)	42.3%	65.3%	39.8%	1.7%	2.5%	10.1%	15.1%
Equatorial (30°)	59.5%	77.8%	84.3%	1.2%	1.7%	6.8%	10.0%

The CPI actually appears to have the best statistics, but uses four satellites, or twice the number of satellites that the Halo does. The next runner up uses six satellites, or three times the number of satellites that the Halo does. The Halo configuration appears to be the best value, using two only satellites. For this reason, it was chosen as the baseline constellation.

CHAPTER 4

Communications Link Analysis

Data passing between Earth and Mars must be placed onto a radio or laser carrier and beamed to the receiver. This carrier must be able to pass data at rates which support user requirements. The data/error rates for various types of service are defined in Table 4-1. The error rate is based upon an assessment of a direct Mars-Earth link for the MESUR program, which specified a Bit Error Rate (BER) of 10^{-7}.[12]

Table 4-1 - Baseline Communications Services/Data Rates

Service	**Data Rate**	**Error Rate (error per bit)**
Low Rate Telemetry	2000 bps	10^{-7}
Medium Rate Telemetry/Voice	64 kbps	10^{-7}
High Rate Telemetry/Video	1 Mbps	10^{-7}
High Rate User Uploads	1 Mbps	10^{-7}

Low Rate Telemetry is described as data such as State of Health (SOH) data, such as vehicle telemetry.

[12]Warren Martin, Anil Kantak, and John Koukos, Mesur Program - An Assessment of a Direct Lander Earth Station (Jet Propulsion Laboratory Study, Dec 1991), 6

Medium Rate Telemetry is information such as real-time experiment data, or high speed sensor samples, such as accelerometer or attitude sensor data. This service could also provide digitized voice capability for manned missions.

High Rate Telemetry would support high speed dumps of stored experiment data. This service would also support video transmissions, and high rate payload data.

High Rate User Uploads are defined as database updates, Morale, Welfare, and Recreation (MWR) library updates, and academic course updates for manned missions.

For the purposes of this study, there are two classes of user considered: 1) a series of small science landers sending data back to Earth; and 2) a large inhabited science station requiring full communications (video, voice, and data). The small lander (MESUR class) will require approximately 1 Mb per Sol (Martian solar day) to be transferred to earth. The large station will require periodic two way video, high speed/high volume data dumps, and MWR uploads.

In determining the properties of these communications links, several factors were examined. A sample mix of data services was defined, and the required data rates were calculated. These data rates were used as the basis for the link analysis calculations. These sample services, which are the author's estimate of what might be required, are detailed in Table 4-2.

Table 4-2 - Sample Communication Services and Data Rates (kbps)					
Communications Services	**Data Rate**	**Earth to Relay**	**Relay to Mars**	**Mars to Relay**	**Relay to Earth**
Relay Telemetry (Tlm)	2 kbps	N/A	N/A	N/A	**2**
Relay Commands	2 kbps	**2**	N/A	N/A	N/A
Low Rate User Tlm (30)	2 kbps	N/A	N/A	**60**	**60**
Medium Rate User Tlm (5)	64 kbps	**320**	**320**	**320**	**320**
High Rate User Tlm (2)	1 Mbps	N/A	N/A	**2000**	**2000**
User Commands (40)	1 kbps	**40**	**40**	N/A	N/A
High Rate User Uploads (1)	1 Mbps	**1000**	**1000**	N/A	N/A
Totals (kbps)		**1362**	**1360**	**2380**	**2382**

The numbers in column one in parentheses represent the number of channels in each type (i.e. 30 channels of low rate user telemetry, 5 channels of medium rate telemetry, etc.). These numbers are estimates, based upon the projected mission requirements. As the Totals show, if each service were scheduled full-time, the system will require over 2 Mbps on the Mars-to-Earth link, and over 1 Mbps on the Earth-to-Mars link. It is likely that these services will need to be timeshared in order to fit into available capacity, but, for now, they act as a target for this analysis' calculations.

Terms Defined

Antenna Gain

An antenna is usually designed to "amplify" an input signal. It accomplishes this by focusing the output power into a narrow beam (rather than allowing it to radiate in all directions), or collecting an incoming signal across a large area and focusing it onto a smaller receiver aperture. Gain is defined as the increase/decrease in transmitted or received power due to the focusing of the communications signal by the antenna. It is specified as a ratio of the antenna output power to that seen from an isotropic radiator (one which radiates equally in all directions), or as the ratio of the signal power output to the receiver (P_o) to the received signal power incident (P_i). It is usually expressed in decibels (dB), defined as 10 log (P_o/P_i). Expressing gain in this fashion allows communications link calculations to be performed by adding gains and subtracting losses, rather than multiplying and dividing, making the calculations much simpler.

Bit Error Rate

Bit Error Rate (BER) is another important parameter in communications link calculations. It is defined as the probability of an error occurring in a bit stream, and is usually specified as the probability per bit of an error (e.g. 1×10^{-5}, which means one error per 100,000 bits). This parameter is the primary driver of the link equations.

The mission requirements specify a tolerable error rate in data transmission (typically 10^{-7}, one error per ten million bits). This error rate will drive a minimum value for the link margin.

Link Margin

Link Margin defines how strong the received signal must be above noise to successfully decode the signal, and is usually specified in dB (a 3 dB link margin would require that the received signal be at least twice as strong as the received noise {3 dB equals a power ratio of 2 to 1}). The value varies with the equipment used and the method used to encode the information on the carrier, and can range from approximately 4 to 10 or higher. In the following calculations, 10^{-7} was used as the required BER[13].

The encoding method used throughout these calculations was Binary Phase Shift Keying (BPSK) with R-1/2 Veterbi decoding, which requires a minimum link margin of 5.7 dB[14] to achieve a BER of 10^{-7}. BPSK with Veterbi encoding provides the lowest signal to noise ratio to achieve a desired bit error rate, at the cost of increased overhead on the link due to the repeated symbols used in the error correction encoding.

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[13]Martin, Kantak, and Koukos, 6

[14]Wiley J. Larson and James R. Wertz, Space Mission Analysis and Design, 2nd ed.(Torrance, CA: Microcosm Inc.) 530

Noise Temperature

Noise Temperature is another important factor in calculating the link margin. Noise Temperature is a parameter which is used to determine the amount of noise input to the receiver. The higher the noise temperature, the more noise entering the receiver, the lower the Signal to Noise ratio of the signal, and the more errors introduced to the signal. In the calculations below, the author made the following estimates of noise temperatures: relay looking into deep space - 100 K; relay looking near Sun (no closer than one or two degrees to the Sun, outside the main beam of the antenna) - 300 K; relay looking at the surface of Mars - 400 K.

Link Analyses

In all of the following calculations, two cases were calculated: 1) a small, unmanned science lander using a 3 dB hemispherical antenna (which simplifies the lander antenna pointing requirements); and 2) a staffed science station (could also define this as a large autonomous science lander) using a 2m transmit/receive dish.

The following equations were used to calculate the graphs below (the quantities in parentheses indicate that the value of that parameter is a function of those quantities, defined in Table 4-3):

Table 4-3 - Definitions of Symbols used in Link Analysis Equations	
Symbol	Definition
R	Range
λ	Wavelength
T	Noise Temperature
B	Bandwidth
D_R	Diameter of Receive Antenna
D_T	Diameter of Transmit Antenna
f	Frequency
c	Speed of Light
L_S	Space Loss
L_T	Loss in Transmitter
L_R	Loss in Receiver
G	Antenna Gain

Equations Used

Eq 4-1. Signal to Noise ratio (SNR) equals received signal power (P_R) divided by noise power (N). In dB terms, the equation becomes P_R - N.

$$SNR(R,\lambda,T,B,D_R,D_T)=P_R(R,\lambda,D_R,D_T)-N(T,B) \quad (4-1)$$

Eq 4-2. L_S is the Space Loss, or the loss in signal strength caused by traveling from point A to point B, and converted into dB.

$$L_s(R,\lambda)=20\log(\frac{4\pi R}{\lambda})(4\text{-}2)$$

Eq 4-3. Power Received equals Power Transmitted minus the loss in the Transmitter, plus the Gain from the transmitting antenna, minus the Space Loss as the signal travels to its destination, plus the Gain of the receiving antenna, minus the losses in the Receiver.

$$P_R(R,\lambda,D_R,D_T)=P_T-L_T+G(D_T,\frac{c}{\lambda})-L_s(R,\lambda)+G(D_R,\frac{c}{\lambda})-L_R(4\text{-}3)$$

Eq 4-4. The Gain of an antenna is a function of the Diameter of the antenna (D), the frequency it is operating at (this formula takes the frequency in GHz - denoted "f_{GHz}"), and its efficiency (η).

$$G(D,f)=20.4+20\log(D)+20\log(f_{GHz})+10\log(\eta)(4\text{-}4)$$

Eq 4-5. The Noise a system generates (n) is related to its Noise Temperature, and the Bandwidth of the data being passed through it (B). It is then converted into decibels (Eq 4-6) for use in the other equations.

$$n(T,B)=kTB(4\text{-}5) \qquad N(T,B)=10\log(n(T,B))(4\text{-}6)$$

Relay to Earth/Earth to Relay

In the Earth-to-Relay case, the calculations were performed using the maximum distance between Earth and Mars, 377 million km. No attempt was made to run different sets of calculations for the Halo Constellation, which lies approximately one million kilometers from Mars, as the differences in the distances from the three points to Earth (376 million, 377 million, and 378 million km) were so small as to be insignificant (0.2 percent).

Our goal for this link is to have the ability to support approximately 2.4 Mbps from the relay to Earth, and approximately 1 Mbps from Earth to the relay, as shown in Table 4-2.

Figure 1 shows the Signal-to-Noise Ratio (SNR) vs Link Data Rate for several frequencies, using a 5 m parabolic transmit dish on the relay and a 15 m parabolic receive dish at the ground station on Earth with a receiver noise temperature of 100 K. The horizontal line in the graph is the required SNR, which is

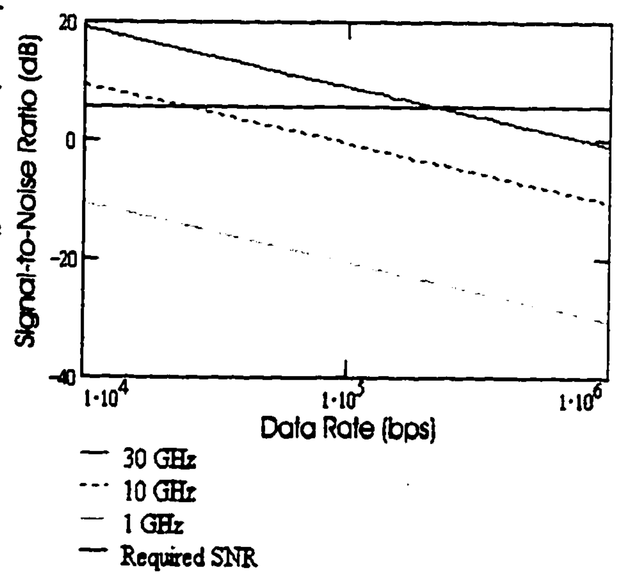

Figure 1 SNR vs. Data Rate (Families of Transmit Frequency). 5 m transmit antenna on Relay, 30 W transmitter, 15 m receive dish on Earth, 100 K Noise Temp.

set at 5.7 dB (based on the selected coding method and BER) for the purposes of these calculations.

Using a transmit frequency of 30 GHz, a 5 m transmit antenna, and a 15 m receive antenna, the link can support rates approaching 300 kbps, which is short of our goal. Using 10 GHz would reduce the link capacity by a factor of nearly 10, to approximately 30 kbps, and 1 GHz, far below that. Based on this information, a link frequency of 30 GHz was chosen.

It may be noted that the slope of the lines decreases as the data rate increases. This is caused by the fact that, as the number of bits per second (frequency) is increased, the amount of energy in each bit is decreased. The ratio, then, of signal energy to noise energy will decrease.

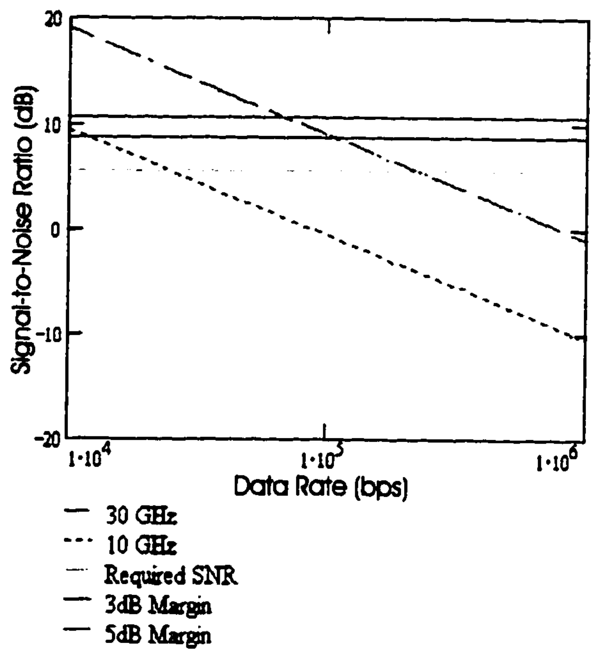

Figure 2 SNR vs. Data Rate. This graph illustrates the effect of adding link margin to the link equations. The bottom horizontal line represents the link capacity with no additional margin. The middle horizontal line shows a 3 dB margin, and the top horizontal line, 5 dB.

Figure 2 illustrates the effect of adding increased margin into the link calculations. Figure 1 uses the required SNR (5.7 dB for the particular type of modulation and encoding {BPSK, R-1/2, K=7, Viterbi Soft Decoding} used in these calculations.) to determine the "floor" for how much data can be passed over the link. This does not take into account variations in the received signal strength caused by natural and human-made noise sources, antenna pointing inaccuracies, or other causes. Figure 2 adds in a 3 dB and a 5 dB "margin" to account for such variations, raising the "Required SNR" to make the link by that amount. In this way, a data rate chosen by referring to the 3 dB line can sustain a 3 dB increase in noise, or decrease in signal strength, and still maintain the link. A data rate chosen using the "Required SNR" line in Figure 1 cannot. The result of adding this margin is to decrease the data throughput, with the size of the decrease dependent upon how much safety margin the user desires. This reduction is approximately 40% for a 3 dB margin, and 60% for a 5 dB margin. The reduction in data rate will occur in the same ratio throughout this chapter, and thus will not be shown in following graphs. The data rates described below are stated with no margin included, as the margin allowed is an operational choice made by the user.

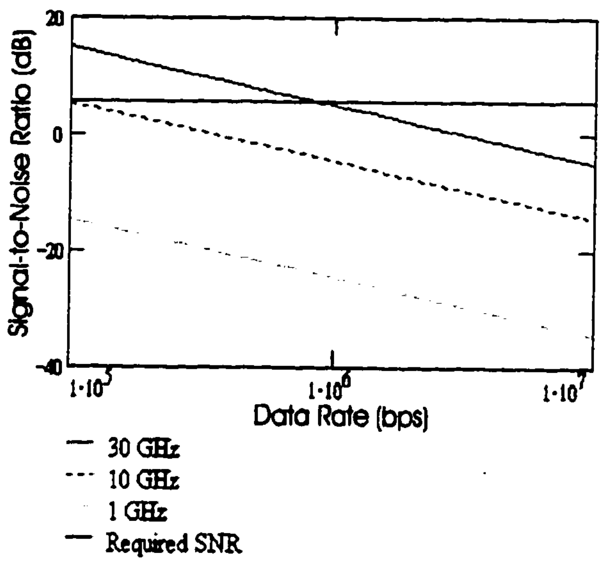

Figure 3 SNR vs. Data Rate (Families of Transmit Frequency). 10 m transmit antenna on Relay, 30 W transmitter, 15 m receive antenna on Earth, 100 K Noise Temp.

Figure 3 displays the result of increasing the receive antenna size (on Earth) to 30 m. This change increases the link capacity to nearly 1 Mbps.

Figure 4 shows the result if the antenna sizes were switched, doubling the size of the antenna on the Relay instead of at the ground station (10 m on Relay, 15 m on Earth). As expected, Figure 3 and Figure 2 show the same result, because each has one antenna doubled in size from the first case.

Figure 5 shows a 10 m transmit antenna on the Relay and a 30 m receive antenna on Earth. At 30 GHz, the link will support approximately 5 Mbps from the

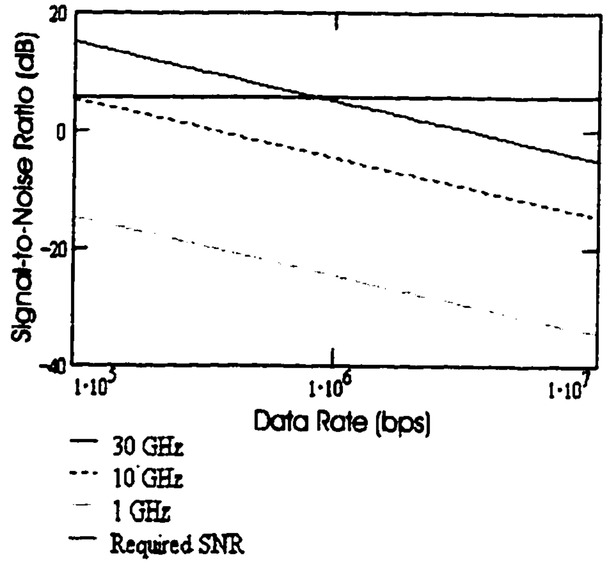

Figure 4 SNR vs. Data Rate (Families of Transmit Frequency). 5 m transmit antenna on Relay, 30 W transmitter, 30 m receive antenna on Earth, 100 K Noise Temp.

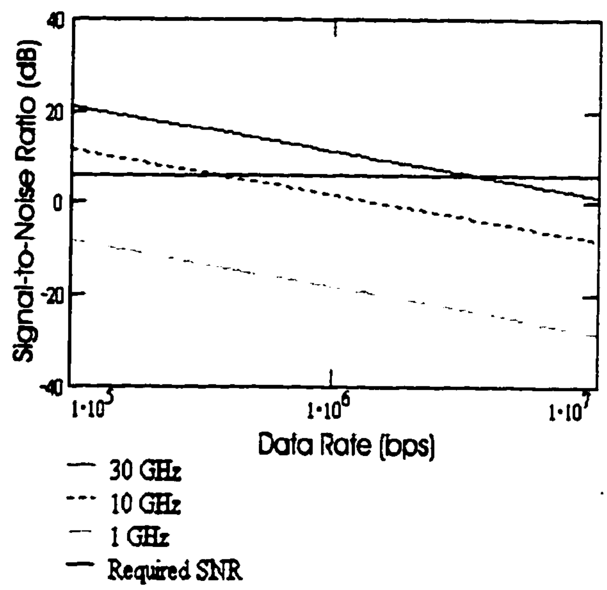

Figure 5 SNR vs. Data Rate (Families of Transmit Frequency). 10 m transmit antenna on Relay, 30 W transmitter, 30 m receive antenna on Earth, 100 K Noise Temp.

Relay to Earth (looking out into deep space). This data rate exceeds our goal of 2.4 Mbps.

The previous graphs illustrate data rate calculations without the Sun in the line-of-sight path. As the communication path line of sight comes closer to the Sun, more of the noise generated by the Sun will enter the antenna, degrading the received signal. The degraded signal will force the data rate lower (which will increase the SNR), until an adequate SNR is reached. It is quite possible that during this period, the full mission data rate will be unachievable. The following calculations use a Noise Temperature value of 300 K, to model the additional radio frequency noise generated by the Sun.

As seen in Figure 6 (compare with Figure 1), the combination of a 5 m transmit dish on the Relay, and a 15 m receive dish on Earth provides a data throughput of approximately 10^5 bps, which is a reduction of a factor of nearly three compared with Figure 1. This reduction will grow as the Line-of-Sight path nears the Sun.

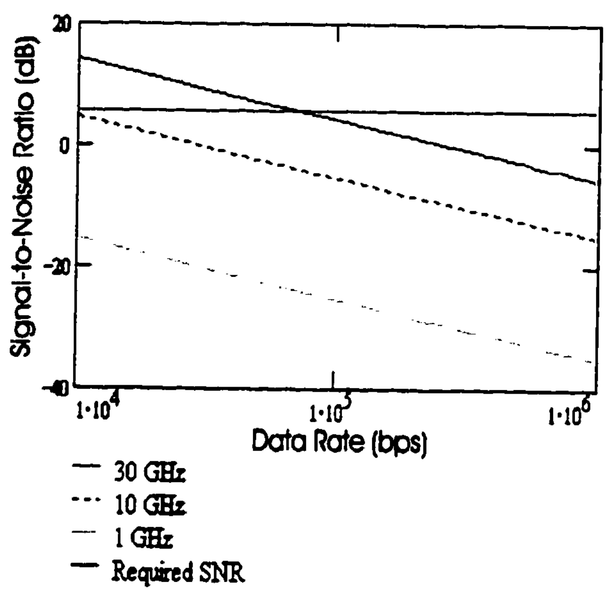

Figure 6 SNR vs. Data Rate (Families of Transmit Frequency). 5 m transmit antenna on Relay, 30 W transmitter, 15 m receive antenna on Earth, 300 K Noise Temp.

Figure 7 (compare with Figure 3) increases the Earth antenna side of the link to 30 m. Predictably, it increases the data throughput by a factor of five, equivalent to the increase seen in Figure 3.

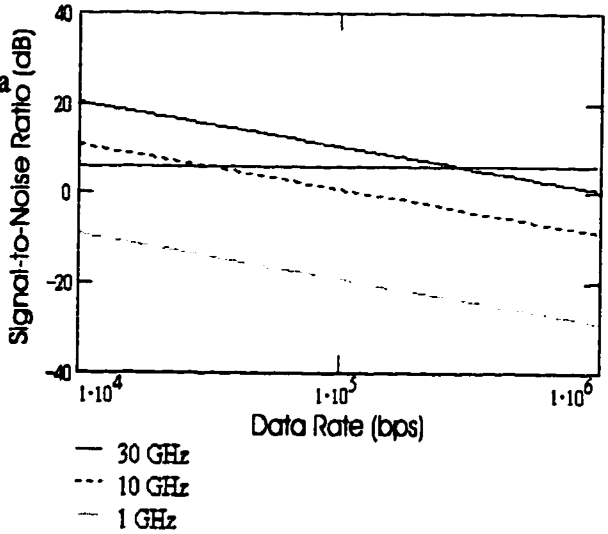

Figure 7 SNR vs. Data Rate (Families of Transmit Frequency). 5 m transmit antenna on Relay, 30 W transmitter, 30 m receive antenna on Earth, 300 K Noise Temp.

Figure 8 shows an increase similar to that in Figure 7 (and proportional to the increase seen in Figure 4), as the larger antenna has merely changed sides of the equation and the communications link.

Figure 9 (compare with Figure 5), shows once again that by using the largest antennas, the best performance for a given output power is attained.

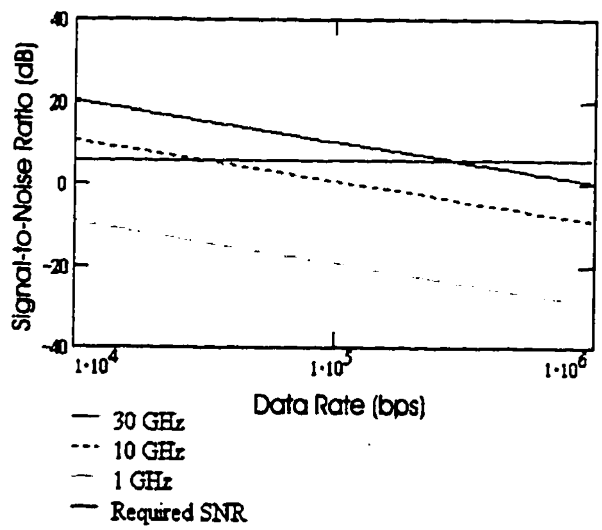

Figure 8 SNR vs. Data Rate (Families of Transmit Frequency). 10 m transmit antenna on Relay, 30 W transmitter, 15 m receive antenna on Earth, 300 K Noise Temp.

These graphs also demonstrate the degradation caused by the Sun as the Line-of-Sight

(LOS) path nears it. As Earth moves closer to the Sun, the Link performance will become worse, due to the increased noise generated by the Sun. Eventually, the performance of the Link will fall far enough that the data rate attempted will need to be reduced before a successful link can be achieved. These cases should be rare, occurring for about two weeks every 700 days or so, as the Earth - Mars Line-of-Sight approaches the Sun. There will also be a several day period during which the link will be unavailable, as the Sun passes directly between the Relay and the Earth. During this period, all Mars customers will be out of touch with Earth for several days (hence, there would be a requirement on the

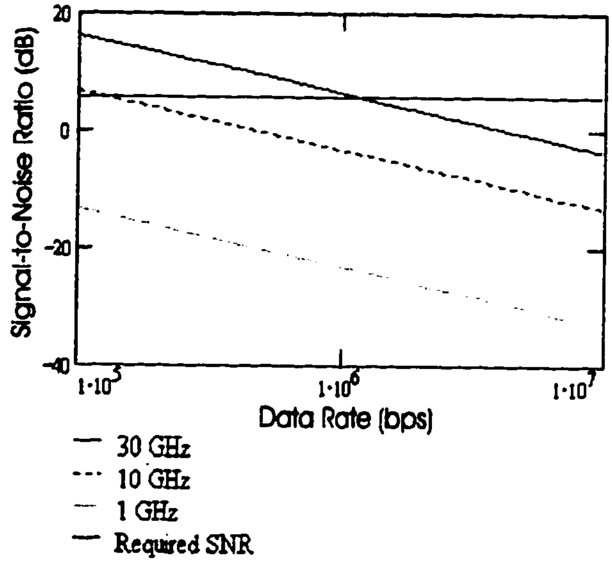

Figure 9 SNR vs. Data Rate (Families of Transmit Frequencies). 10 m transmit antenna on Relay, 30 W transmitter, 30 m receive antenna on Earth, 300 K Noise Temp.

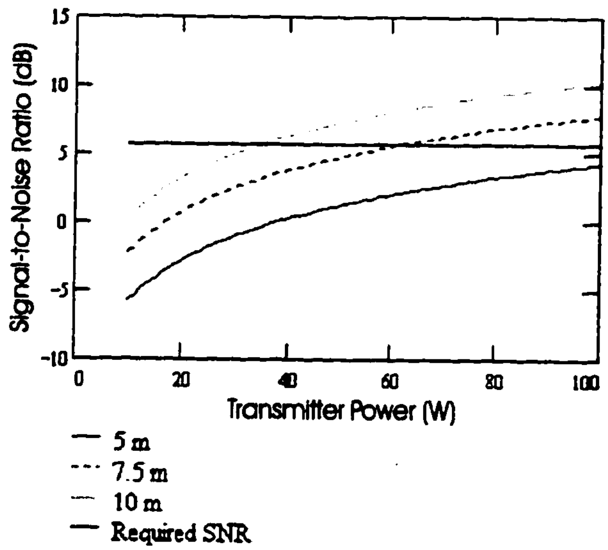

Figure 10 SNR vs. Transmitter Power (Families of Relay Antenna size). Assumes 15 m receive antenna size on Earth, 1 Mbps data rate, and 100 K Noise Temp.

spacecraft/landers to be able to operate autonomously for a period of several days). Examining the graph of Relay transmitter power vs SNR for a given link, Figure 10 shows that a 15 m receive antenna on Earth could support a data rate of 1 Mbps, given various antenna sizes and output power levels. Note that to support 1 Mbps with a 5 m relay antenna would require an output power level of approximately 110 W. Therefore, a 5 m transmit/15 m receive combination will not support 1 Mbps using a reasonable transmit power level.

If the receive antenna on Earth is increased in size to 30 m, the resulting performance is shown in Figure 11. As the graph

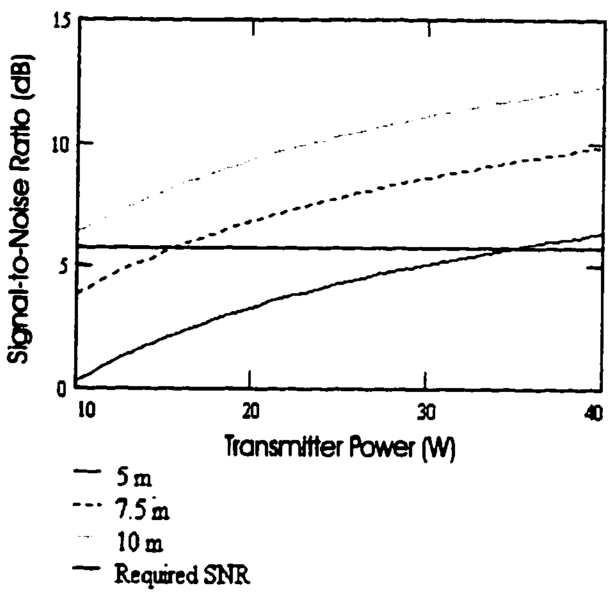

Figure 11 SNR vs. Transmitter Power (Families of Relay Antenna size). Assumes 30 m receive antenna on Earth, 1 Mbps data rate, and 100 K Noise Temp.

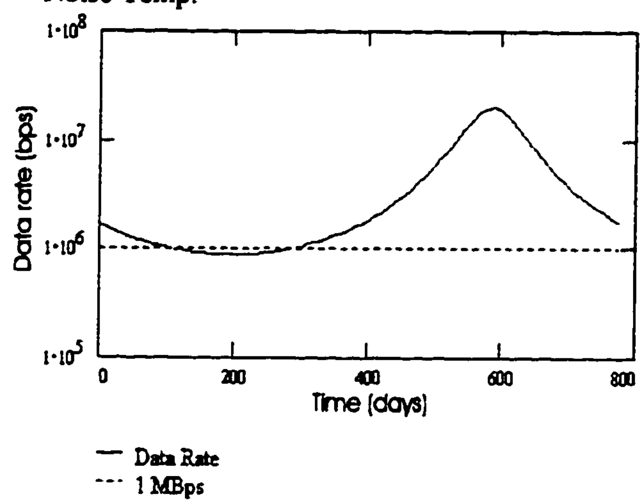

Figure 12 Data Rate vs. Time over one synodic period. This graph illustrates the variation of data rate caused by varying distance between Earth and Mars over the course of their orbits. This plot does not take into account the increase in noise temperature as the Earth - Mars line-of-sight nears the Sun.

shows, using a 30 m receive antenna allows the transmit antenna size to be reduced to 5 m, and the required power reduced to 35 W. Again, let it be stated that these figures are for the maximum Earth - Mars distance, and will improve as the two planets come closer together during their orbits.

As the planets approach one another, the distance between them will decrease, reducing the loss between the transmitter and receiver due to distance. Figures 12-16 illustrate how the data rate will vary over the course of an orbit, falling below 1 Mbps as the LOS path approaches the Sun, and increasing again as the LOS path moves away from the Sun. Figure 12 illustrates the variation over the course of the entire orbit. It does not take into account the increase in noise temperature as the LOS path.

Figure 13 presents an estimate of the amount of time required for the line-of-sight path to move through a given number of degrees (i.e. if communication cannot be supported within two degrees of the Sun, the

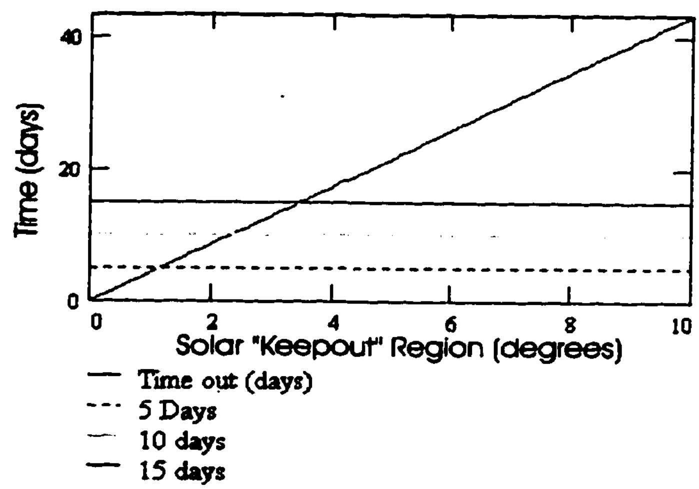

Figure 13 Time (days) vs. Distance from the Sun (degrees). This illustrates the time required for the Earth - Mars LOS to move across the Sun, given a certain "keepout" distance. Inside this "keepout" region, noise generated by the Sun will overwhelm the Earth - Mars communication link. This graph is also used to determine the time period to use the "Near Sun Noise Temp," given an angular distance from the Sun.

line will be out for approx. 8 days). It also illustrates how long to use the "Near Sun Noise Temperature," rather than the deep space value (based on switching over at five degrees from the Sun). In reality, this would not be a step function, but a continuous increase in noise. Using these estimates, Figures 14-16 break Figure 12 into three parts: Figure 14 shows the approach to the Sun (from day 0 to approximately day 206), using the "Deep Space Noise Temperature" of 100 K; Figure 15 changes over to the "Near Sun Noise Temperature" of 300 K; lastly, Figure 16 reverts to 100 K as the LOS path moves away from the Sun. Note that there will be a period (not

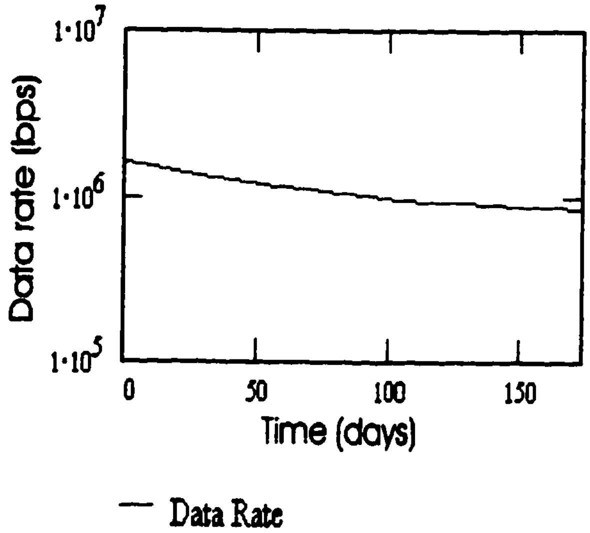

Figure 14 Data Rate (bps) vs. Time (days). Day 0 to day 184, with the LOS approaching the Sun. The data rate will vary with the changing distance between Earth and Mars, which is approaching maximum during the later part of this period.

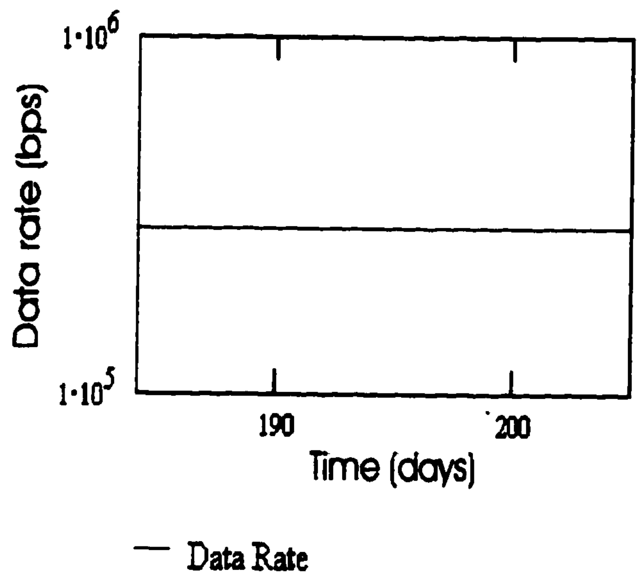

Figure 15 Data Rate (bps) vs. Time (days). Day 184 to day 206, with the Earth - Mars distance at maximum. Using "Near Sun Noise Temp" of 300 K. Does not illustrate the interruption in service as the LOS crosses the Sun.

illustrated in Figure 15) where all communications will be lost as the Earth passes "behind" the Sun, and communications are temporarily lost.

The middle graph (Figure 15) shows the lowest data rate possible for this configuration, is approximately 300 kbps, which will last for a period of approximately 20 days (of which service will be completely unavailable for several days). For the other 760 days per cycle, the achievable data rate is pushed up to nearly 1 Mbps, which is close enough to the original estimate that it can be worked with by scaling either the size or the number of some of the higher rate data links proposed in Tables 4-1 and 4-2.

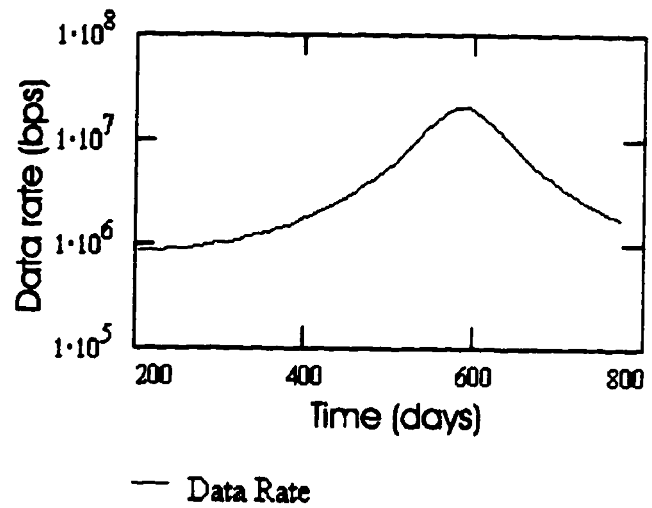

Figure 16 Data Rate vs. Time (days). Day 206 to day 775. At this point, the LOS has moved far enough away from the Sun to begin using the "Deep Space Noise Temp" again. Earth - Mars distance decreases from maximum through minimum. The data rate increases up to approximately 20 Mbps at closest approach.

<u>Summary.</u> The above graphs illustrate that a 35 W transmitter at 30 GHz, utilizing a 5 m transmit antenna and a 30 m receive antenna on Earth, will not adequately carry the data rates postulated in Table 4-2. It will, however, carry enough capacity that all the postulated services could be supported, on a time-shared basis, over most of the orbit. During the period of time when Sun-Mars-Earth angle is approximately five degrees (in

the region of increasing solar noise), this performance cannot be supported (during a portion of this period, the Earth will pass behind the Sun and no Earth - Mars communications will be possible).

Mars to Halo/Halo to Mars

The Mars to Halo and Halo to Mars cases were run using a maximum distance of one million kilometers between the Relay and the ground user.

The goal for this link, for a large station, is to be able to support a total data rate of approximately 2.4 Mbps from the surface of Mars to the Relay, and approximately 1 Mbps from Relay to the surface of Mars, as shown in Table 4-2. The goal for a small station, a subset of Table 4-2, is to be able to support one command link (1 kbps from relay to lander), and one telemetry uplink (1 kbps from lander to Relay).

Downlink. A Relay transmit antenna of 1.8 m in diameter was chosen for the Relay-to-Mars link (halo) at a frequency of 30 GHz. With this size antenna, an entire hemisphere of the planet could be covered in its main beam. While a larger antenna could have been used for the downlink, it would have focused the beam more tightly, and not achieved hemispheric coverage.

From the Relay to the surface of the planet, the goal stated above was to be able to support a small science package with a 1 kbps command downlink and a 1 kbps telemetry uplink. The goal for a "staffed" science station is to be able to supply a 1.3 Mbps command/data link from the Relay to/from the user.

Figure 17 shows the results for a small lander with a 3 dB, hemispherical receive antenna, a 20 W antenna on the Relay, and a 500 K Noise Temp. The graph shows SNR vs Data Rate for various transmit frequencies (with the chosen frequency, 30 GHz, compared day to night), with the horizontal line representing the minimum 5.7 dB SNR required to sustain a 10^{-7} BER.

Figure 18 shows results for a staffed ground station on Mars. This station uses a 2 m antenna for communicating with the relay. This

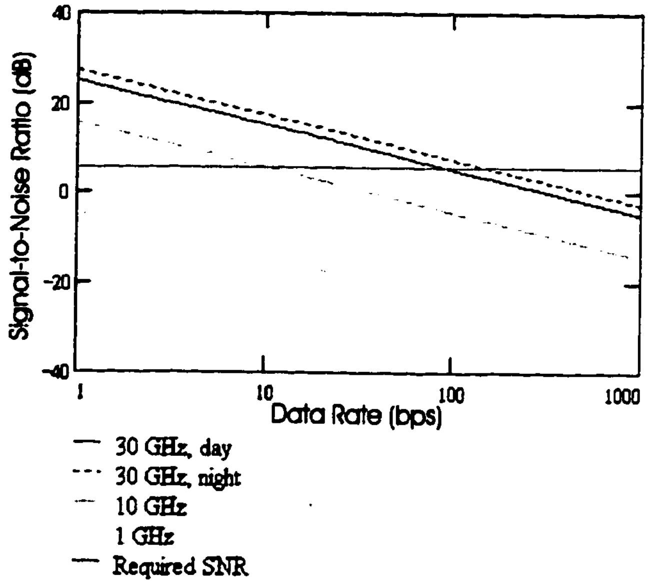

Figure 17 Data Rate vs. SNR (Families of Transmitter Frequency). Small lander using hemispherical (3 dB) antenna. 20 W transmitter on Relay, 30 GHz, 500 K Noise Temperature (night 300K)

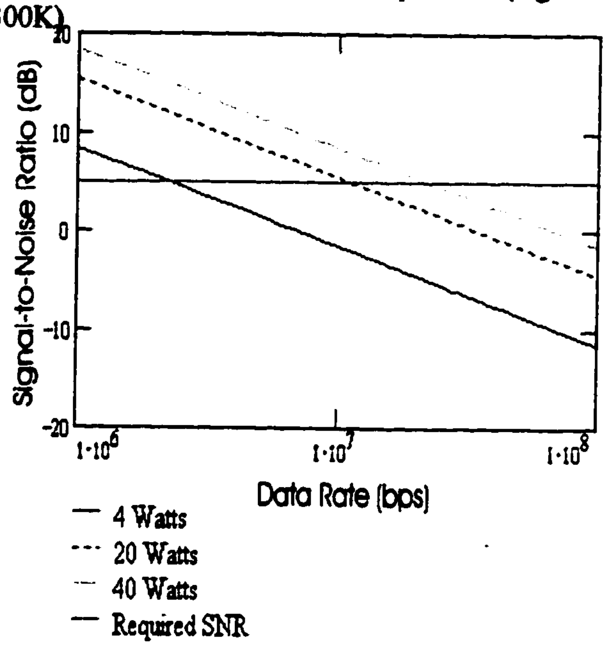

Figure 18 SNR vs. Data Rate (Families of Transmitter Power). Staffed Science Station. 500 K Noise Temp, 30 GHz transmit frequency.

allows a much higher rate of communication through the Relay (by a factor of nearly 10^5). By increasing the ground antenna size, the data rate increased to nearly 10 Mbps, which greatly exceeds the capability of the Relay to Earth link. This extra capacity could be used for inter-relay transfers (transferring data between ground stations on opposite sides of the planet).

Figure 19 shows the relationship between transmitter power and data rate, based on a 30 GHz transmit frequency, for a small science lander.

This configuration can pass approximately 200 Bps between the Relay and lander. This is

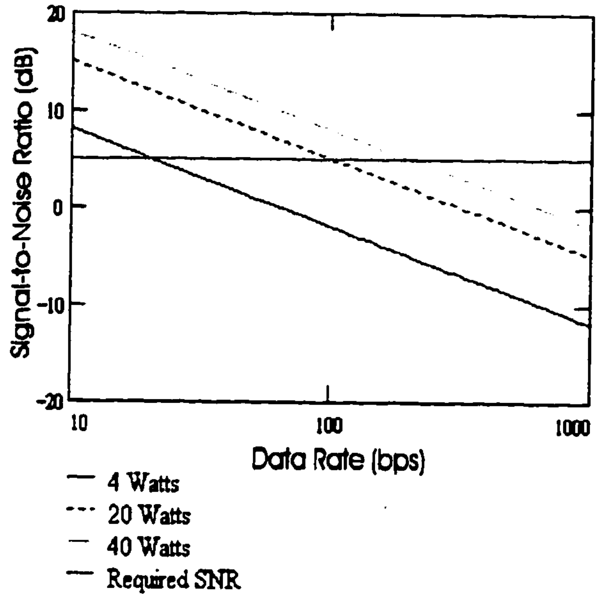

Figure 19 SNR vs. Data Rate (Families of Transmitter Power). 500 K Noise Temp, 30 GHz transmit frequency.

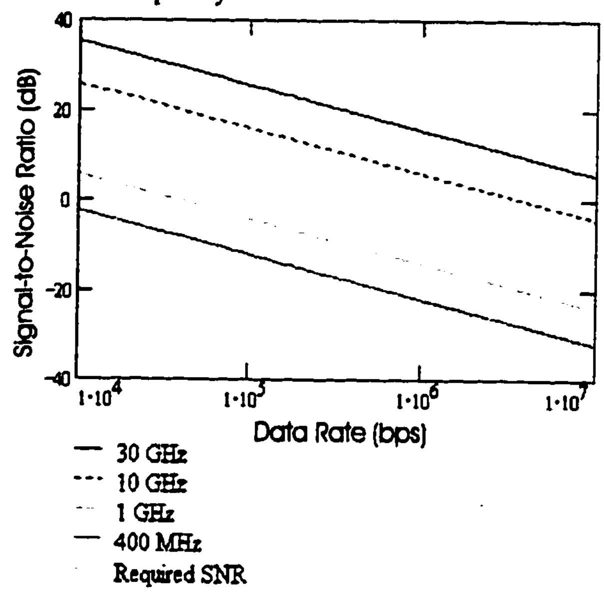

Figure 20 SNR vs. Data Rate (Families of Transmit Frequency). Large ground station using 2 m antenna. 500 K Noise Temp, 20 W transmitter on Relay.

approximately 10 times the capacity currently available, and is adequate for commanding a small science lander.

Figure 20 shows a similar graph for the large station (2 m antenna). Using a 20 W transmitter on the Relay, the downlink can support a data rate of approximately 10 Mbps.

To summarize, the data indicates that a 20 W transmitter and a 1.8 m relay antenna is required in order to support a downlink rate of 200 Bps to a small lander, and 10 Mbps to a large station. 10 Mbps is somewhat higher than what is required, but the extra capacity could be reduced by using a smaller antenna on the ground.

Uplink. Figure 21 shows that a 4 W transmitter on a small science station can support an uplink data rate of approximately 30 bps. This rate will support the transmission of approximately 2.6 Mb in a 24 hr period.

Moving to the large station (Figure 22), the same 4 W transmitter will support

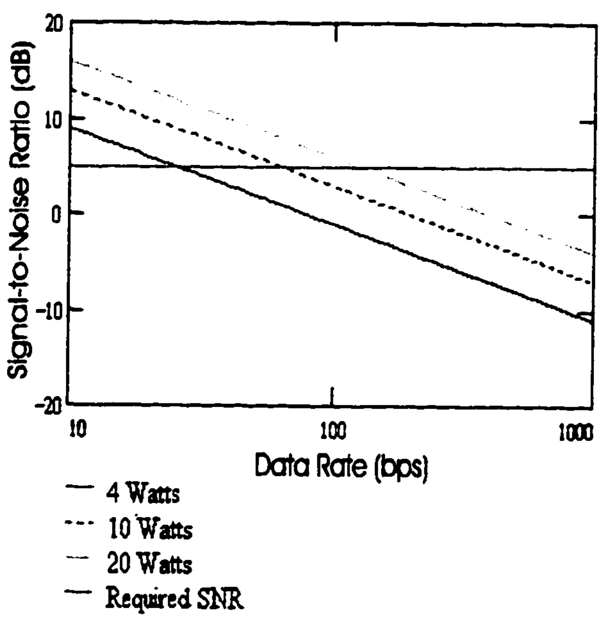

Figure 21 SNR vs. Data Rate (Families of Transmitter Power). Small Lander. 400 K Noise Temp, 30 GHz transmit frequency.

approximately 2.5 Mbps, moving up to 10 Mbps using a 10 W transmitter. This data rate should be adequate for such a station, and exceeds the capacity of the Relay to Earth link.

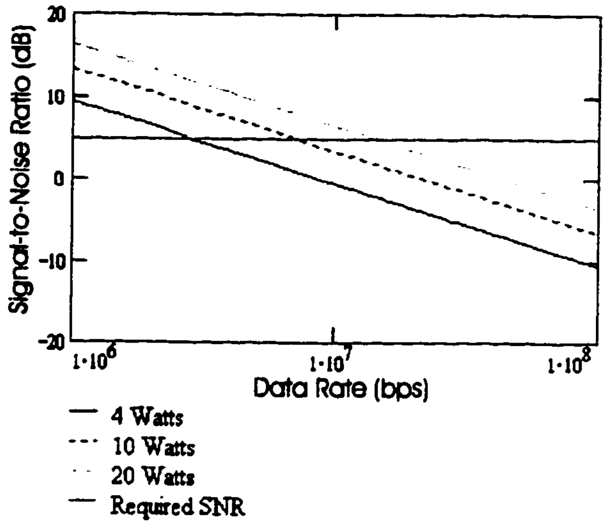

Figure 22 SNR vs. Data Rate (Families of Transmitter Power). Large Science Station. 400 K Noise Temp, 30 GHz transmit frequency.

In summary, a 1.8 m parabolic antenna on the Relay was selected to provide complete coverage of Mars' surface. A 20 W transmitter/amplifier combination will support a 200 bps downlink from the Relay to a small science lander. The same 20 W output will support a 10 Mbps downlink to a large science station, which far exceeds the capability of the Relay-Earth link. This does suggest, though, that the additional capacity could be utilized by routing small station transmissions down to a large station *and* send the data back to Earth, which would allow the scientists on Mars (at the large station) to make use of the data collected, as well as get it back to Earth for their colleagues. A small lander can achieve a 30 bps rate using a 4 W transmitter through a hemispherical antenna, for a total of approximately 2.6 Mb per day (the requirement is 1 Mb per day). The same 4 W through the large station's 2 m dish would provide a 2.5 Mbps link, which far exceeds the capability of the Relay-Earth link.

Mars to Aerosynchronous/Aerosynchronous to Mars

The Aerosynchronous case data was calculated using a distance equal to the maximum distance from an aerosynchronous satellite to the edge of the visible horizon, approximately 20,180 km. It, like the other cases, is designed to support a small science lander with a 1 kbps telemetry uplink and a 1 kbps command downlink, and a "staffed" science station with a 1.3 Mbps up/down link capability. In this case, however, the configuration uses a horn antenna instead of a parabolic dish to support the link to the surface. A small dish would also be able to deliver the signal to the ground also, but in order to spread the signal the required 18.4 degrees (the angular width of Mars from aerosynchronous altitude), the diameter required for the dish would be very small, on the order of 10 cm (4 in) in diameter. An antenna that small would cause a great deal of scattering of energy, and send only a small portion of the transmitted signal toward the planet. It is for this reason that a horn antenna was chosen for the aerosynchronous link.

A 15 GHz transmit frequency was chosen for a similar reason. Given that a horn

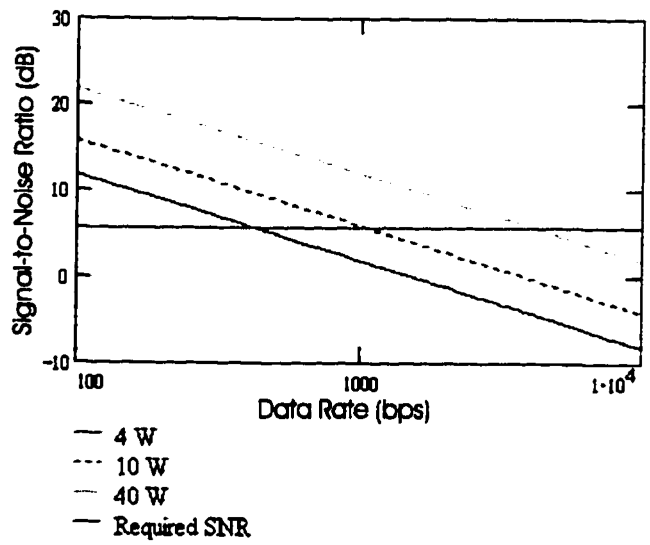

Figure 23 SNR vs. Data rate (Families of Transmitter Power). 500 K Noise Temp, 15 GHz transmit frequency, 3 dB receive antenna.

antenna was to be used, the 15 GHz frequency allowed the horn to be of a manageable size.

Downlink. Figure 23 displays the relationship between Signal-to-Noise Ratio and Data Rate for families of transmitter power, for a small science lander on the day side of Mars. A 10 W transmitter aboard the Relay will support approximately a 1 kbps data rate to a small lander, which is a large increase over current capabilities, and much greater than required for a command downlink to a small lander. A 4 W transmitter on the Relay will support approximately a 1 kbps downlink to the small station, which is sufficient for a command link.

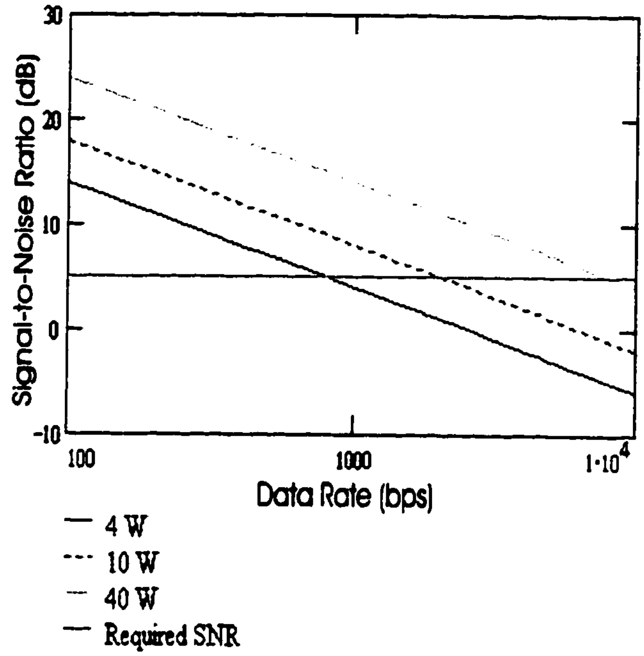

Figure 24 SNR vs. Data rate (Families of Transmitter Power. 300 K Noise Temp, 15 GHz transmit frequency, 3 dB receive antenna.

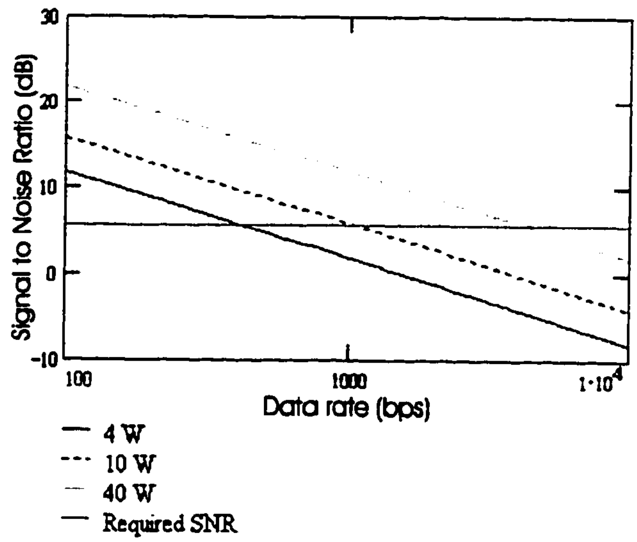

Figure 25 SNR vs. Data rate (Families of Transmitter Power). 500 K Noise Temp, 15 GHz transmit frequency, 3 dB receive antenna at small lander.

Figure 24 illustrates the same case for the night side of Mars.

Figure 25 displays the result of increasing the Noise Temperature to 500 K, depicting the increased noise as the Relay nears the Sun. The same 4 W transmitter will now support approximately 600 bps, which remains sufficient for our needs. It must be noted, however, that there will be short periods, during the Equinoxes, when the Relay will pass directly in front of the Sun. During these periods, the downlink from the Relay will become lost in the solar noise, and the user will be unable to command the lander.

Figure 26 shows results for the "large" ground station, using the 2 m receive dish

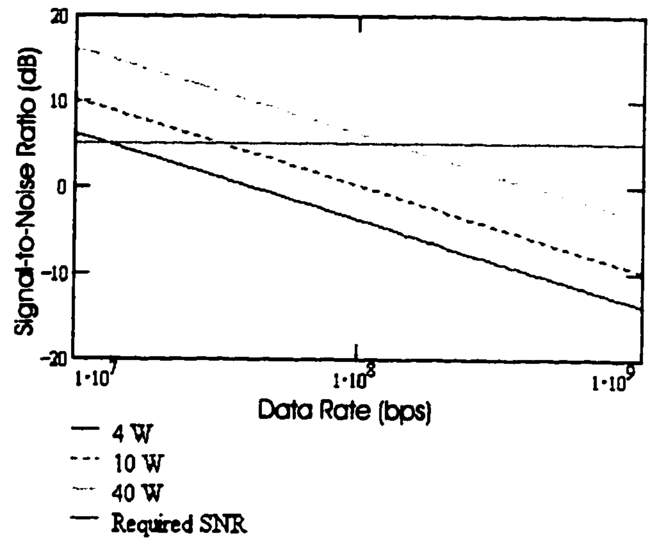

Figure 26 SNR vs. Data Rate (Families of Transmitter Power). 15 GHz, 2 m parabolic receive antenna at large station. 500 K Noise Temp.

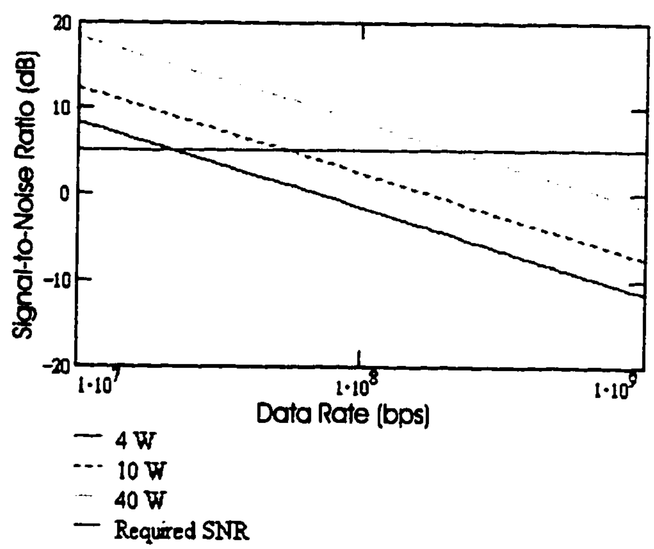

Figure 27 SNR vs. Data Rate (Families of Transmitter Power). 15 GHz, 2 m parabolic receive antenna at large station. 300 K Noise Temp.

mentioned previously. Because the Relay is so much closer to Mars than in the last case, the 4 W transmitter is now capable of sending data at above 10 Mbps, which exceeds the capacity of the Relay-Earth link. Figure 27 shows similar data, but looking away from the Sun, into "deep space." The increase in data rate due to the decrease in noise is approximately 2:1.

Uplink. Figure 28 shows the data rate a small lander can achieve using its 3 dB hemispherical antenna. Here, the Relay is now looking at Mars, so the Noise Temperature has been switched to 400 K. In this configuration, using a 4 W transmitter, the lander can uplink approximately 700 Bps, or nearly 60 MB per day, which is far in excess of our requirement of 1 MB per day.

In Figure 29, the results for a large station show a similar capability. A 4 W transmitter can uplink in approximately 15 Mbps through the station's 2 m dish. Again, this is far in excess of what the Relay is capable of sending back to Earth.

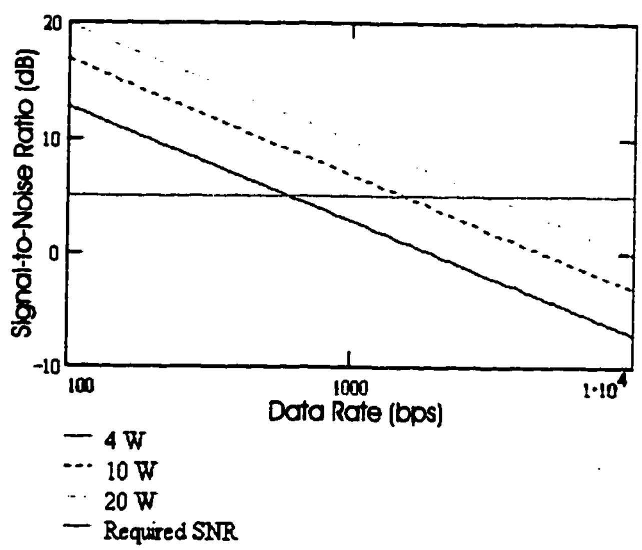

Figure 28 SNR vs. Data Rate (Families of Transmitter Power) for small science lander. 15 GHz transmit frequency, 3 dB transmit antenna, 400 K "Mars" Noise Temp.

As shown above, a lander using a 4 W transmitter can uplink data at rates in excess of the requirement laid on it in Table 4-2. The downlink will exceed Table 4-2 requirements during most of the Mars year, though will encounter some difficulty during the equinoxes, when the Sun will pass directly behind the Relay. During these periods, which could last from one to two hours, the user will be unable to command the lander. The staffed station will suffer a similar effect, though likely for a shorter period, as its more focused antenna rejects more solar noise as the Relay approaches the Sun. This degradation is a product of this particular type of orbit, and is commonly seen on terrestrial communications satellites. It is a predictable occurrence, and is typically scheduled in to the daily operations plan as a scheduled outage. There should be no impact to lander operation, as long as the daily operations plan is uploaded to the lander prior to the outage.

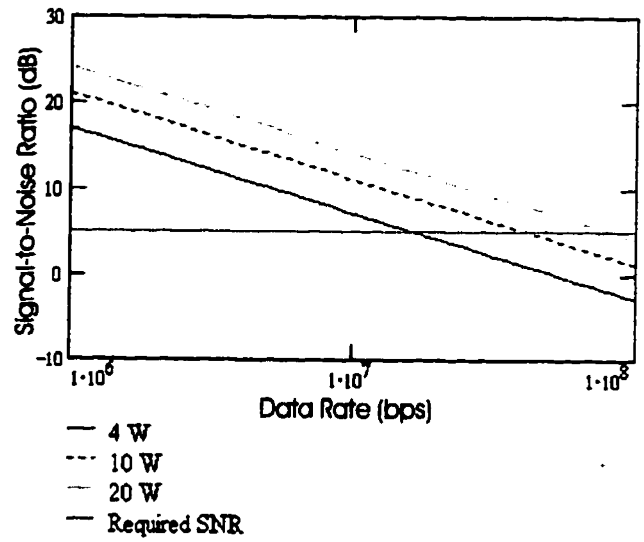

Figure 29 SNR vs. Data Rate (Families of Transmitter Power) for a large science station. 15 GHz transmit frequency, 2 m transmit dish, 400 K Noise Temp.

Mars to Low or Medium Orbit/Low or Medium Orbit to Mars

The Low/Medium Orbit case calculations were made using a distance equal to that from the Relay satellite to the edge of the visible horizon, approximately 1472 km for the Low case, and approximately 2805 km for the Medium orbit case.

Two orbits were chosen to be representative of this configuration: a 308 km "Low" orbit (13 revs/orbits per Sol); and a 1015 km "Medium" orbit (10 revs/orbits per Sol) (more specific information on these orbit types is discussed in Chapter 3, Constellation Configuration). The orbit selections were not "optimized" to provide the best coverage distributions. The low and medium cases are grouped together because of the similarity in the calculations for both cases, with only slight differences in the magnitude of the critical parameters.

A Helix antenna was chosen for this configuration. A Horn antenna could not be practically (in terms of its physical size) made to spread the signal widely enough to support the Low/Medium orbit

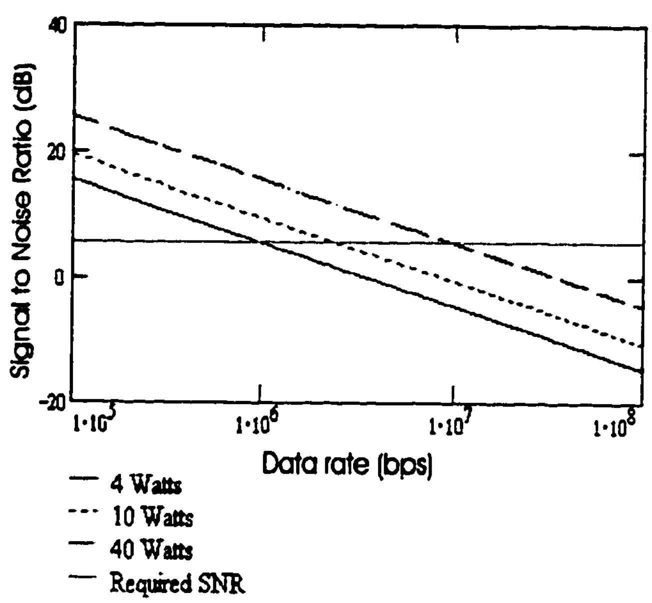

Figure 30 SNR vs. Data Rate (Families of Transmitter Power). Low Altitude, 3 dB receive antenna on small lander, 300 K (night) Noise Temperature, 400 MHz transmitter frequency.

configuration, nor could a parabolic dish. The operational frequency was changed to 400 MHz as well, as a consequence if changing the antenna design. Lastly, the Relay transmitter power output was reduced to 4 W from 20 W. The Relay is much closer to the user than in the other cases, and, as illustrated in Figure 30, 4 W proved adequate to carry the data rates required.

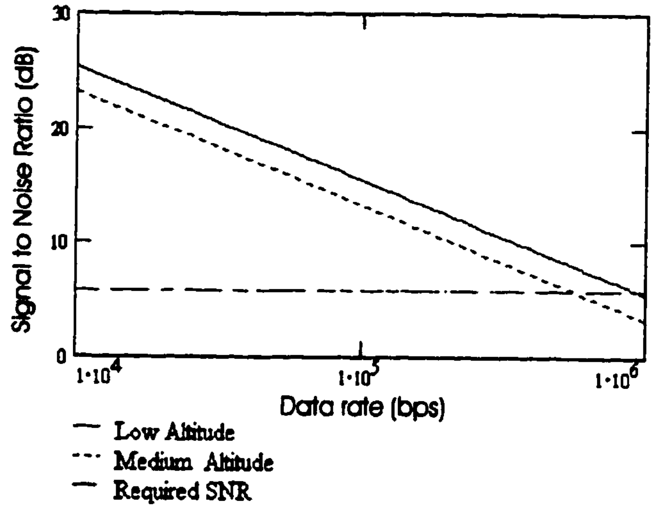

Figure 31 SNR vs. Data rate (Families of Orbiter Altitude). 4 W Transmitter on Relay. 300 K Noise Temp (night), 400 MHz transmit frequency, 3 dB receive antenna.

<u>Downlink.</u> Because the Relay is closer to the lander, the amount of data passed is greatly increased. Figure 31 illustrates the relationship between Signal-to-Noise ratio and Data Rate for the small science lander to relays

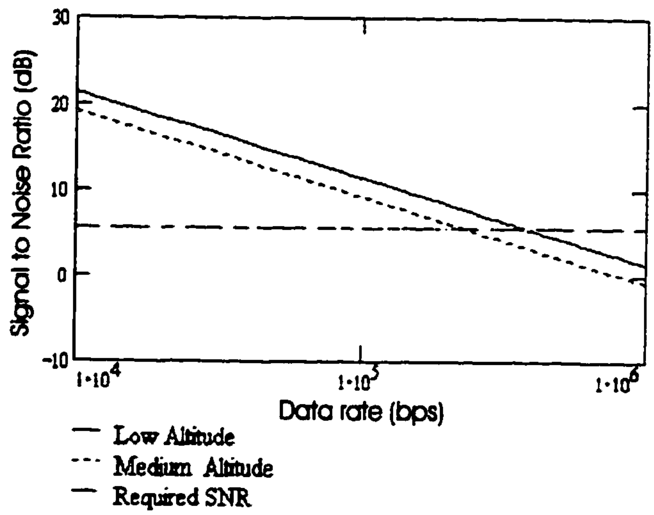

Figure 32 SNR vs. Data rate (Families of Orbiter Altitude). 4 W Transmitter on Relay. 500 K Noise Temp (day), 400 MHz transmit frequency, 3 dB receive antenna.

at Low and Medium altitude, at night (Noise Temperature 300K). As stated above, these graphs were calculated using a 4 W transmitter onboard the Relay (the great reduction in altitude from the other cases allowed this change). As shown in the graph, the differences in the downlink are significant (approximately 3 dB, or a factor of two), but not greatly different.

Figure 32 shows a similar graph, but for the day sky (Noise Temperature 500K). The relative difference of approximately 3 dB remains, although the absolute magnitude drops by nearly 6 dB (a factor of approximately four to five), due to the increased noise generated by the warmer daytime atmosphere. Both of these graphs show a marked increase in the ability to transfer data to the small lander (on the order of a factor of thirty), as compared to the other, higher altitude cases. This, again, is due to the fact that, in this case, the Relay is so much closer to the lander than before.

The next two figures show the same data for the staffed station. Figure 33 illustrates the night-side numbers, using a 2 m receive antenna and a 300 K night-time noise temperature, a 4 W transmitter,

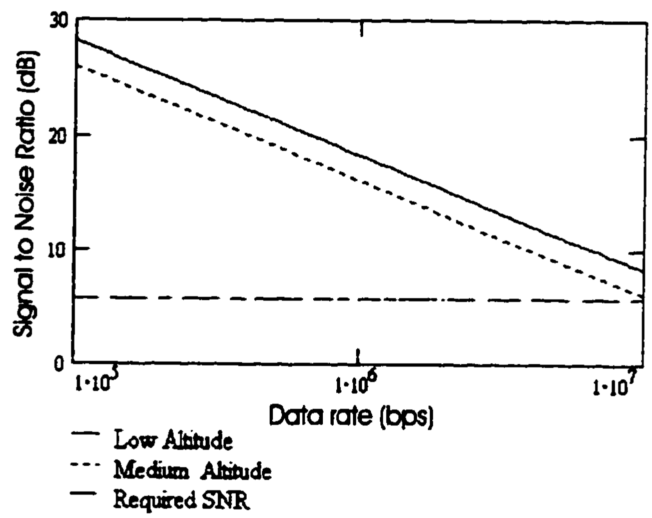

Figure 33 SNR vs. Data rate (Families of Orbiter Altitude). 4 W Transmitter on Relay. 300 K Noise Temp (night), 400 MHz transmit frequency, 2 m receive antenna at Large Station.

and a 400 MHz transmit frequency. Figure 34 illustrates the day-side results (using a 500 K noise temperature). Though the relative difference between the two orbit types remains (as one would expect), the magnitude of the downlink has increased greatly from the small lander case. However, in comparison to

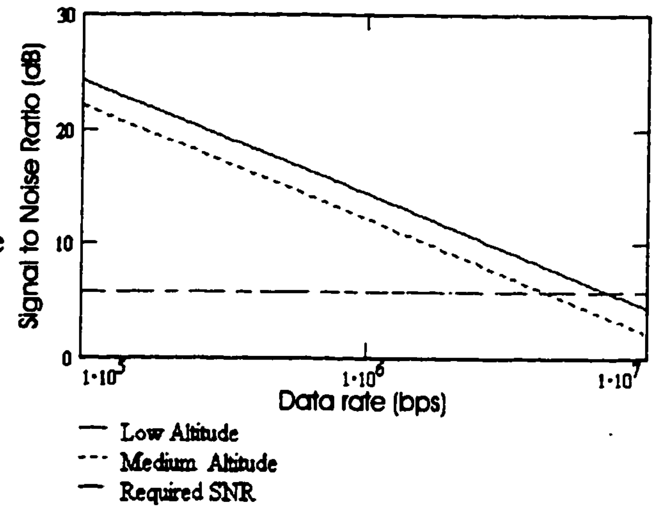

Figure 34 SNR vs. Data rate (Families of Orbiter Altitude). 4 W Transmitter on Relay. 500 K Noise Temp (day), 400 MHz transmit frequency, 2 m receive antenna at Large Station.

Figure 24 (Aerosynchronous case), for instance, the throughput is nearly the same (near 10^7 for a 4 W output), though one would expect an increase similar to those seen for the small lander. This is explained by pointing out that, though the Relay is closer (and hence has a stronger signal at the receiving antenna), antenna gain is a function of frequency. By reducing the operating frequency to 400 MHz, the gain of the 2 m parabolic antenna was greatly reduced (it is more efficient at higher frequencies), to the point where the increase created by the reduced distance was balanced by the loss caused by the reduced operating frequency. Regardless, the data rate capability is still in excess of 1 Mbps, and is more than the Relay to Earth portion of the link can carry.

Summary. Given the greatly decreased range of this configuration, it is possible to reduce the power output of the relay from 20 W, as seen in previous configurations, to 4 W and maintain a downlink data rate which is more than adequate, and which far exceeds the capability of the Earth-Relay link to support. The staffed science station, due to the reduction in operating frequency and Relay output power, did not see a proportional increase in its throughput rate, though the previous levels were maintained. It is important to note here that the increased downlink rates could be required due to the shorter "in contact" times available to a user in this configuration. The amount of time the station will have available to receive data downlinked to it will vary greatly on successive orbits. In some cases, this could be as little as three minutes or less, increasing to approximately fifteen minutes or more. This is caused by the fact that a satellite will pass within view, though not directly over, a ground site on several successive orbits. The variation in contact times is caused by the varying elevation of the satellite at the ground station. This variation will require mission schedulers to take into account which "pass" they schedule particular data/mission downlinks for, to ensure that there is adequate time for the user to receive and verify all information before the end of the contact.

Uplink. The uplink from the small station showed an increase similar to that of the downlink, for the same reasons. Again, increased data rates are necessary in this configuration, given the much shorter contact times. Even so, Figure 35 illustrates a capability for the large station to operate in excess of 1 Mbps, even using a 1 W transmitter through its 2 m antenna. Since this is nearly the maximum possible data throughput for the Relay - Earth link, using a 1 W transmitter in this case would be an acceptable solution. If, however, the users requirements called for passing large amounts of data between several large stations, increasing the output to 4 W (which is not a terribly significant change in terms of weight or station power budget) would allow the extra capacity for a "dump and store" type capability aboard the Relay. This would be useful in this particular configuration, given that there

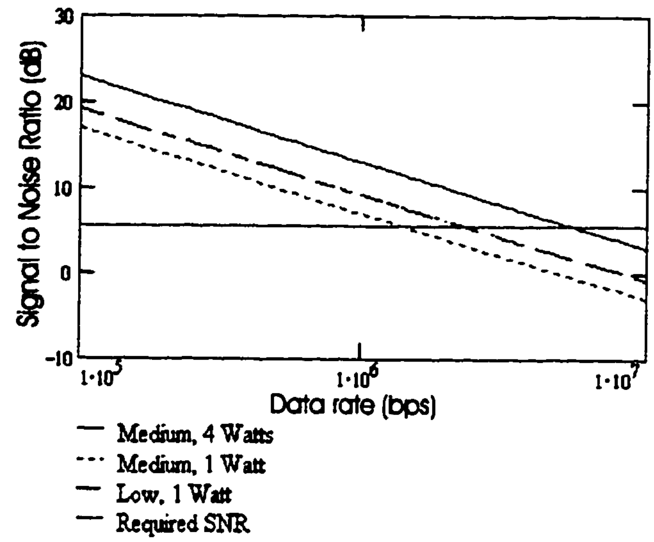

Figure 35 SNR vs. Data Rate (Families of Transmitter Power). Large science station, 2 m transmit antenna, 400 MHz transmit frequency, 400 K Noise Temp.

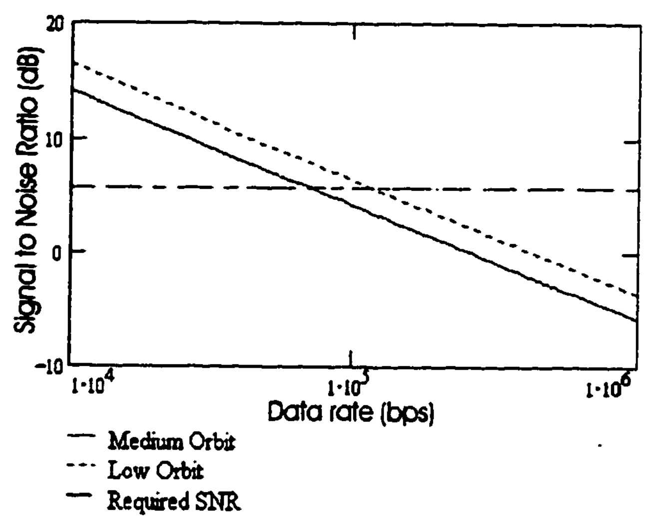

Figure 36 SNR vs. Data Rate (Families of Transmitter Power). Small science station, 3 dB transmit antenna, 400 MHz transmit frequency, 400 K Noise Temp.

would be no satellite visible to more than one station at a time, as in the higher altitude cases.

Figure 36 illustrates the small lander case. Again, the graph illustrates results using a 1 W transmitter, as there did not seem to be a need for a large "dump and store" capability. Again, the higher data rate is required due to the shorter contact times. Even so, if the data transmission overhead were an incredibly high 80% (80% of the symbols transmitted are used for link synchronization, packet identification, etc, as opposed to actual data), uplinking the entire 1 Mbps per day requirement would take 100 seconds per day (or two contacts of 50 seconds each, per day), which is well within the time budget for this configuration.

Summary. A 1 W transmitter provides both classes of ground station with sufficient capacity to meet its data transmission requirements. In addition, if a "store and dump" capability for communication between staffed science stations were desired (which would probably provide the only "direct" means of station to station communication), it could be accommodated by increasing the large station's transmitter power to 4 W. This small change would not have a large impact on the station's weight or power budget.

Mars to Common Period, Inclined (CPI) Orbit/ CPI Orbit to Mars

All calculations for the Common Period, Inclined (CPI) orbit class were made with the Relay at apoapsis (the most distant point in the orbit), or what should be the maximum range from the ground user. No attempt was made to model the system dynamics and determine if another point actually had a greater range than that at apoapsis. The maximum range was calculated as the maximum slant range from a user on the edge of the Relay Field of View. This method models the theoretical "worst case" range.

The CPI constellation involves four satellites in elliptical, inclined orbits. Two of the satellites have apoapsis in the northern hemisphere, and two in the south. Because the Relays are in elliptical orbits, they can remain over their respective hemispheres for most of their orbit, as the Relay will move more slowly near apoapsis (just as a ball you toss up in the air slows down and stops, then begins to fall back to your hand). Because the Relay is changing altitude, there will be an "operational" period in its orbit, as well as a "non-operational" one (as it moves through periapsis, or the closest point in the orbit to the planet).

A horn antenna was chosen for this configuration, for much the same reasons that it was for the aerosynchronous case, in that a parabolic dish designed to cover the entire surface of the planet would be very small and would greatly scatter the transmitted/received energy.

The operating frequency choice was not the same as for the aerosynchronous case, however. Because the altitude of the Relay varies in this case, the notional design included a lower gain, less focused helix antenna for communication during the "non-operational" portion of the orbit (near periapsis). The operating frequency was chosen to be compatible with both antennas, and was chosen as 2 GHz.

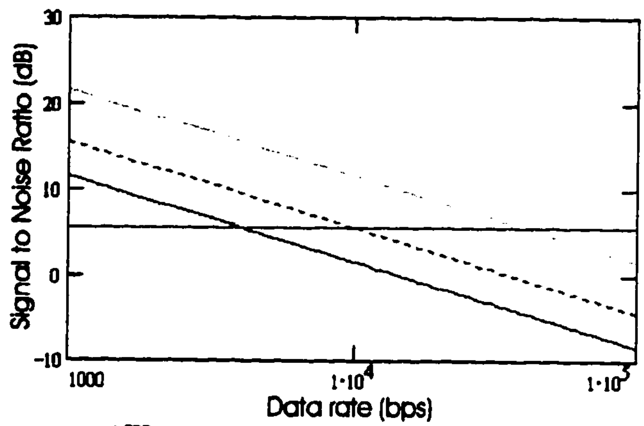

Figure 37 SNR vs. Data rate (Families of Transmitter Power), 300 K Noise Temp, 2 GHz transmit frequency, 3 dB receive antenna at small lander.

<u>Downlink.</u> The following graphs depict the downlink performance of the Relay to both the small science station and the large staffed station. Figure 37 illustrates the small lander receiving data from the Relay on the night side of the planet. As illustrated in the figure, the data link can achieve a rate on the order of 5-6 kbps using a 4 W transmitter aboard the Relay. This is far in excess of our 1 kbps requirement for a command/data uplink to a small lander.

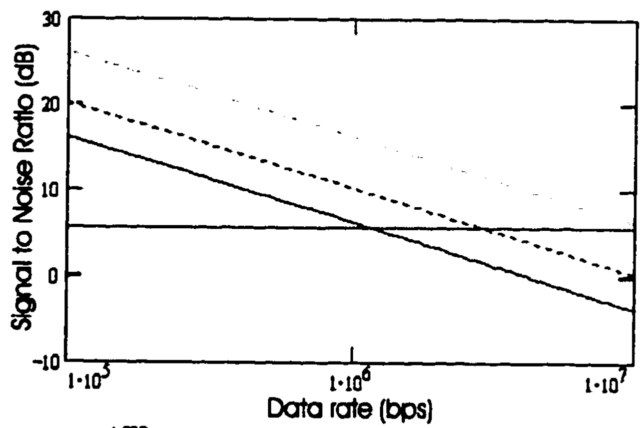

Figure 38 SNR vs. Data rate (Families of Transmitter Power), 500 K Noise Temp, 2 GHz transmit frequency, 2 m receive antenna at large station.

Figure 38 provides the same information for the large, staffed station. As with many of the other cases, the calculated link capacity is on the order of 1.2 Mbps, very close to the goal of 1.3 Mbps, using a 4 W transmitter (the higher powered transmitters putting out more data, respectively).

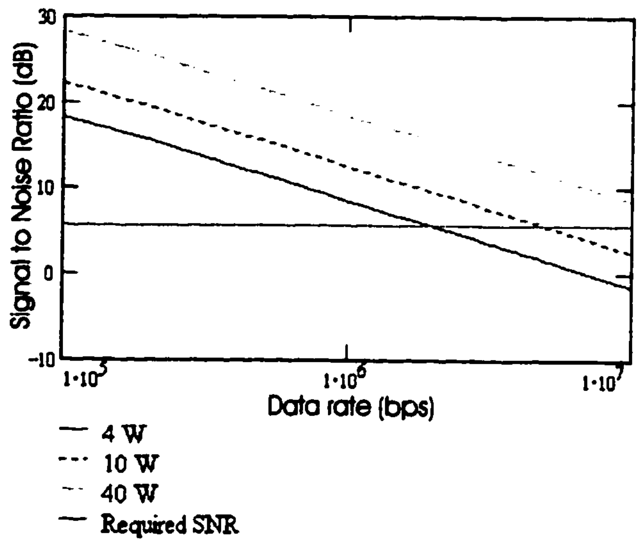

Figure 39 SNR vs. Data rate (Families of Transmitter Power), 300 K Noise Temp, 2 GHz transmit frequency, 2 m receive antenna at large station.

Figure 39 moves the ground station to the day side of Mars, raising the background noise as a consequence (the day sky being warmer and generating more noise, in concert with the RF-noisy Sun being visible, raises the noise level so a lower overall data rate

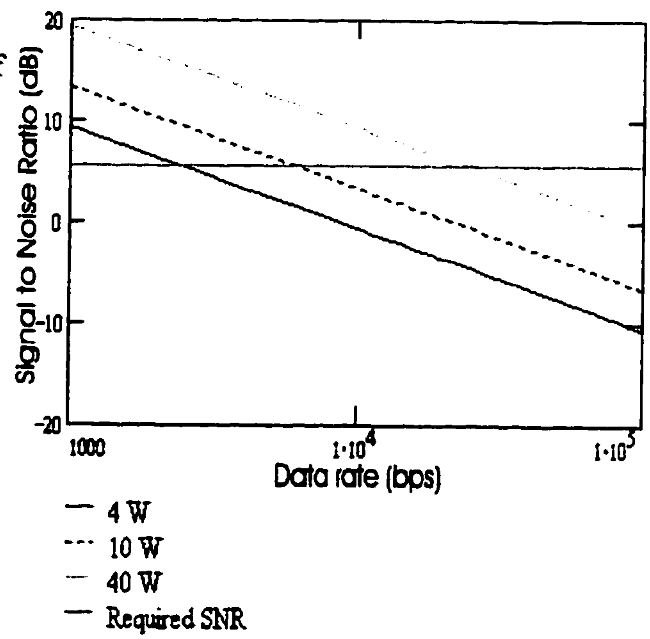

Figure 40 SNR vs. Data rate (Families of Transmitter Power), 500 K Noise Temp, 2 GHz transmit frequency, 3 dB receive antenna at small station.

is achieved). In this case, it achieves slightly over 1 Mbps, or approximately a 10% reduction over Figure 37.

Figure 40 illustrates similar results for the small science station. This case experiences the same reduction in data rate as the larger case did, down to approximately 3 kbps from near 5 in Figure 36. This still provides far in excess of the required rate for a command uplink.

Uplink. The next two figures illustrate the uplink analysis done for this constellation type. As with the previous cases, the small science lander will use a 3 dB "hemispherical" antenna, designed to direct the signal up into the sky, without wasting too much down into the ground. This antenna also eliminates the need to track the satellite, as nearly the whole sky is visible to the antenna. The large station will use its 2 m parabolic dish, which will be able to track the satellite throughout its orbit (given the amount of equipment which would be necessary to a staffed station, a tracking antenna did not seem unreasonable).

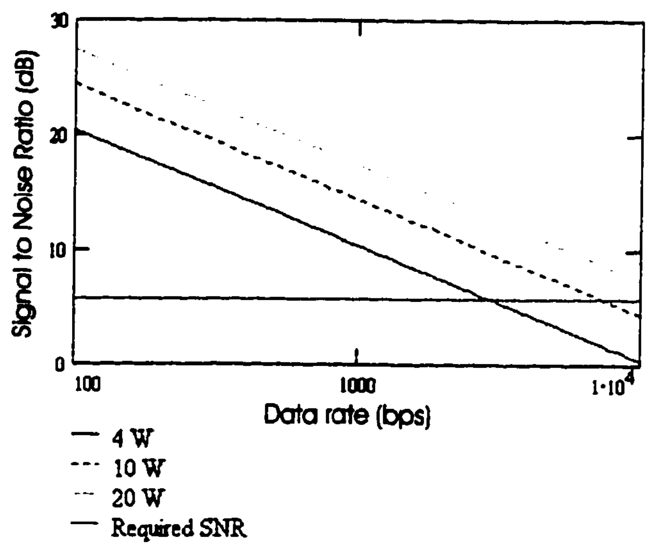

Figure 41 SNR vs. Data rate (Families of Transmitter Power), 400 K Noise Temp, 2 GHz transmit frequency, 3 dB transmit antenna at small station.

Figure 41 illustrates the small lander uplink. The requirement for uplink from a small station is 1 Mbit per day, which can easily be accomplished by the link (at 500 bps, this transfer would take under an hour, and the link capacity exceeds that easily).

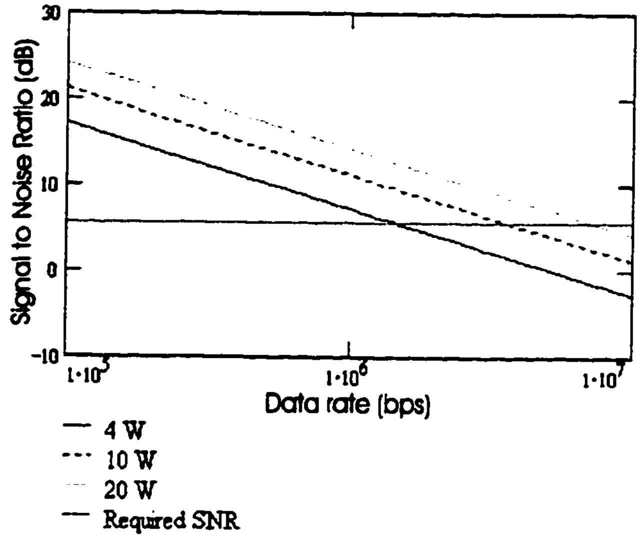

Figure 42 SNR vs. Data rate (Families of Transmitter Power), 400 K Noise Temp, 2 GHz transmit frequency, 2 m transmit antenna at large station.

Figure 42 displays the uplink analysis for the large station. Again,the assessed performance is short of the goal stated in Table 4-2 of 2.3 Mbps, but a 4 W transmitter comes close. If the 2.3 Mbps number is a firm requirement, a larger transmitter can be used to ensure that the requirement is met. The data rates available using 10 and 20 watt transmitters are also illustrated in the graph, and either should provide more than enough capacity to achieve the Table 4-2 goal.

In summary, the Common Period, Inclined constellation compares well to the other constellations in terms of the ability to pass data to and from Earth. The data rates calculated fall close to, if they do not exceed, the estimated mission requirement shown in Table 4-2. The spacecraft and lander configurations used do not require exotic or futuristic technology, only what can be purchased today. Based on the analysis described here, this portion of the mission is feasible using the equipment described here.

Summary.

This chapter has examined the communications capabilities of several constellation configurations to support an Earth to Mars communications network. Based on the requirements laid out at the beginning of the chapter, none of the constellations were able to meet the Mars to Earth "specification" of 2.3 Mbps. That number, however, was arrived at by postulating what services a human presence on Mars might require. Therefore, the specification can be changed to meet what is physically possible. The requirement for High Rate User Telemetry/Video can be dropped from two simultaneous links to one, which will take care of most of the problem. Data Compression should allow the rest of the services to be sent over the links as they are designed. Tables 4-4 through 4-7 summarize the results of this chapter.

Table 4-4 - Summary of Results from Communications Analysis - Small Lander (Downlink)				
Constellation Type	Relay Output Power	Operating Frequency	Day Date Rate	Night Data Rate
Low Orbit	4 Watts	400 MHz	500 kbps	950 kbps
Medium Orbit	4 Watts	400 MHz	350 kbps	650 kbps
Aerosynchronous	4 Watts	15 GHz	700 bps	900 bps
Common Period, Inclined	4 Watts	2 GHz	3000 bps	6000 bps
Halo	20 Watts	30 GHz	100 bps	150 bps

Table 4-5 - Summary of Results from Communications Analysis - Small Lander (Uplink)				
Constellation Type	Relay Output Power	Operating Frequency	Day Date Rate	Night Data Rate
Low Orbit	1 Watt	400 MHz	100 kbps	100 kbps
Medium Orbit	1 Watt	400 MHz	80 kbps	80 kbps
Aerosynchronous	4 Watts	15 GHz	700 bps	700 bps
Common Period, Inclined	4 Watts	2 GHz	3000 bps	3000 bps
Halo	4 Watts	30 GHz	30 bps	30 bps

Table 4-6 - Summary of Results from Communications Analysis - Large Lander (Downlink)				
Constellation Type	Relay Output Power	Operating Frequency	Day Date Rate	Night Data Rate
Low Orbit	4 Watts	400 MHz	8 Mbps	11 Mbps
Medium Orbit	4 Watts	400 MHz	6 Mbps	10 Mbps
Aerosynchronous	4 Watts	15 GHz	13 Mbps	25 Mbps
Common Period, Inclined	4 Watts	2 GHz	1.5 Mbps	3 Mbps
Halo	20 Watts	30 GHz	10 Mbps	10 Mbps

Table 4-7 - Summary of Results from Communications Analysis - Large Lander (Uplink)				
Constellation Type	Relay Output Power	Operating Frequency	Day Date Rate	Night Data Rate
Low Orbit	4 Watts	400 MHz	3 Mbps	3 Mbps
Medium Orbit	4 Watts	400 MHz	1.5 Mbps	1.5 Mbps
Aerosynchronous	4 Watts	15 GHz	15 Mbps	15 Mbps
Common Period, Inclined	4 Watts	2 GHz	1.5 Mbps	1.5 Mbps
Halo	4 Watts	30 GHz	2.5 Mbps	2.5 Mbps

Conclusion.

In this chapter, the communications links for several constellation configurations have been examined. From the data presented, the lower orbiting constellation have the advantage in data rate throughput.

This is as expected, given that the relays are closer to the user. This does not, however, mean that the highest data rate is the best choice.

Given the complexity of the communication switching involved in a low orbiting constellation, these would appear to be less desirable choices for relay satellites, despite the data rate they are capable of carrying. The two best configurations from this standpoint are the CPI and the Halo configurations. Both are capable of passing the daily data rate required by the MESUR mission, and much more to a large science station.

Given those two choices, the data rate throughput may now be compared to determine the best configuration. Overall, the CPI constellation provides the highest data rates. To a small station, the data rate is consistently 30 to 100 times that of the Halo, despite being transmitted with one fifth the power. To a large station, though, the Halo gets the upper hand, due to the fact that a parabolic antenna does not perform as well at 2 GHz as it does at 30 GHz. This limitation, however, could be overcome by selecting an antenna design more optimal for the link.

CHAPTER 5

Summary of the Remainder of this Work

This chapter provides a short summary of the remainder of this work.[15]

Tai incorporates the data presented here with his own work, that of determining the cost modeling equation, calculating the ΔV required to place a Relay satellite in a Libration point orbit (for the Halo Constellation), determining the mass of the Relay spacecraft, so to determine the booster and fuel costs associated with each scenario, determining the constellation monetary cost, the time required to establish the network, and calculating the spacecraft lifetime.

Once all of these parameters are adequately determined, they will be scored and input into the cost modeling equation developed by Tai. This equation will include several sets of coefficient weighting factors, corresponding to several sets of user priorities. The parameter scores will be placed into the equation with each set of coefficients, creating several "least cost" answers corresponding to the constraints imposed by the user.

Based upon the partial data here, and what has been completed to date by Tai, a summary of the parameters determined to date is listed below:

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[15]Tai

Fuel costs are higher to the Halo orbits, due primarily to the much lower velocity the spacecraft is traveling in that orbit, requiring the spacecraft to slow down to a lower velocity to enter the orbit. Stationkeeping fuel costs, though, are much lower for the Halo orbits, again due to the lower velocity the spacecraft is traveling.

Launcher costs are lower to the Libration Point orbits due to the lower number of satellites required to be delivered to the operational orbits.

As mentioned in more detail in Appendix E of this work, the Low and Medium Orbit, Six Satellite Constellations are the most fault tolerant of the constellations examined.

The Halo Constellation requires the least time to construct, given that it has the fewest number of satellites. The Aerosynchronous constellations would be next, due to the fact that all spacecraft are delivered to the same orbit plane.

This preliminary data suggests the following:

For a user who desires performance above all else (communications coverage and communication link coefficients high, others low), the Common Period, Inclined Constellation will likely be the "least cost," due to its high performance figures.

For a user desiring high performance, but limited budget (monetary cost highest, communications coverage and communication link coefficients high), the Halo Constellation will likely be the "least cost," primarily due to the high communications results and the fact that only two satellites are required for high performance.

For a user desiring fault tolerance over all, the Medium Orbit, Six Satellite Constellation configuration would likely be the "least cost," due to its high degree of tolerance to the loss of a spacecraft.

Again, the above are preliminary results. These results will likely change slightly as more of the Trade Study parameters are finalized.

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APPENDIX A

Circular Restricted Three-Body Problem Simplifying Assumptions

As stated in Chapter 4, there is no closed-form general solution to the three-body problem. By making several simplifying assumptions, however, it is possible to determine the locations of five equilibrium points. At these equilibrium points, the gravitational and centrifugal forces acting on a body placed there cancel, allowing the body to remain at or near the equilibrium point with a minimum amount of applied force.

The bodies involved are defined as follows:

1) The Primary body is the one with the largest mass.
2) The Secondary body is the next greatest mass.
3) The Satellite is the body of least mass, and the one whose motion is being studied.

In this work, the Primary and the Secondary are sometimes referred to collectively as the Primaries. As stated above, several simplifying assumptions must be made in order to simplify the equations of motion to the point where one can solve for the equilibrium positions. Those assumptions are as follows:

1) That the Secondary body orbits the Primary in a circular orbit. This allows the positions of the equilibrium points to remain constant. In the elliptical

orbit case, the position of the equilibrium points would vary with the positions of the Primary and Secondary.

2) The mass of the Satellite is assumed infinitesimal as compared to the Primary and the Secondary. This simplification ensures that the orbits of the Primary and Secondary are not affected by the motion of the Satellite.

By making these assumptions, the equations of motion are sufficiently simplified to allow the determination of five equilibrium points. These points, determined by a French mathematician by the name of Lagrange and known as L-points, are located as follows (refer to Figure 1):

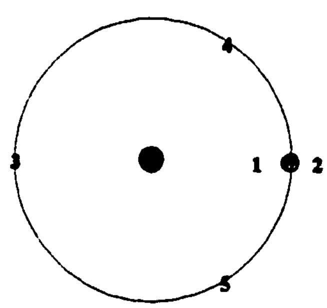

Figure 1 Illustration of the positions of the Lagrange points.

1) L_1 is located along the line connecting the two Primaries and inside the orbit of the Secondary (for the Sun-Mars system, L_1 is located approximately 1 million kilometers from Mars).

2) L_2 is located along the line connecting the two Primaries, and outside the orbit of the Secondary (for the Sun-Mars system, L_2, like L_1, is located approximately 1 million kilometers from Mars, but in the anti-sunward direction).

3) L_3 is located on the opposite side of the Primary from the Secondary.

4) L_4 is located equidistant from the Primary and Secondary, at a distance equal to the distance between the Primary and Secondary (making an equilateral triangle with the Primary and Secondary), and leading the Secondary in its orbit about the Primary.

5) L_5 is located equidistant from the Primary and Secondary, at a distance equal to the distance between the Primary and Secondary (making an equilateral triangle with the Primary and Secondary), and lagging the Secondary in its orbit about the Primary.

This work involves satellites in orbit about L_1 and L_2.

APPENDIX B

Coverage Analysis Program (CVG) Description and Flowchart

The program used to compute the constellation availability (percentage visibility of the constellation around Mars) was written by the author. It takes as input up to six satellite position files, one satellite per file. These are produced either by the program POHOP (developed at the Jet Propulsion Laboratory in Pasadena, CA), used to generate the positions of satellites orbiting Mars, or a program written by the author (referred to as HALO), which calculates the position and velocity of a body in orbit about the Sun-Mars Lagrange points (L_1 or L_2) by integrating the Circular Restricted Three-Body Problem (CR3BP) equations of motion (which are described in more detail in Chapter 4). CVG reads the files one line at a time, each line giving the position and velocity of the satellite at a specific time. Each file must cover the same time period, in the same size time steps (e.g. each satellite's position must be "observed" at the same time). As CVG reads a line from a file, it compares the time to the time read from the first file. If they do not agree, CVG prints a warning message and terminates.

Assuming the times of each satellite's "observation" agree, CVG reads each satellite's position information. POHOP provides the X, Y, and Z position in Mars-Centered-Inertial (MCI) coordinates, while HALO outputs data in the coordinates of a

Rotating Reference Frame, centered at the Sun-Mars barycenter and rotating with the Sun-Mars system (again, these are described further in Chapter 4). If the file contains halo orbit data, the program translates each line of data from the rotating frame to the MCI frame. Once all satellite vectors have been converted (if required), CVG "loops" around the surface of planet, calculating the elevation of each satellite as seen from each latitude/longitude point on the surface. At each point, the program stores the maximum elevation reached by any one of the satellites in the constellation.

Three matrices have been allocated (and initialized to zero) to store the visibility data. If the maximum elevation at the current point is greater than zero degrees, CVG increments the corresponding element of the first matrix (visibility at zero degrees or better). If over ten degrees, CVG increments the second, and if over thirty degrees, the third. This, in effect, counts the number of times during the simulation that at least one satellite in the constellation was visible at that point on the "surface" (there is no count of whether more than one satellite was visible, though this could be done with minor modifications to the code). By also counting the number of "timesteps" in the run (visibility opportunities), CVG determines the percentage time that a ground point had access to the constellation by dividing each element of the three matrices by the number of timesteps. The resulting matrices contain percent availability, and are output in a comma-delimited, ASCII text file. These matrices are then read into Mathcad® 4.0 for Windows® for analysis. Mathcad® is used to generate color contour charts of the data, allowing easier analysis of the data. These contour charts are shown in Chapter 4.

There is some special formatting of the output files performed by CVG due to the method that Mathcad® uses to generate the contour charts. Mathcad® reads the data file, and assigns colors in the chart based on the range of data it "sees". In order to get consistent color mapping between cases, it was necessary to insert four "range points" (since the possible values in the matrices are between zero and one, two "0" values and two "1" values are inserted) in order to ensure that the range of values in the data files would be consistent (a portion of the analysis was performed by using only every other point to generate the contour chart [this greatly sped up the chart generation, while not greatly impacting the accuracy of the output]). Four points ensured that two ranging points were included, even if only every other point was used.

Given that CVG uses rectangular matrices to store data regarding a simulated sphere, the data files have some redundancy in them, namely multiple points which all represent the North and South Poles. Because of this, it was possible to use two of these redundant points to insert the "range points." These points are not used in any of the scoring calculations performed on the data, but merely to ensure that the color mapping done by Mathcad® would be consistent.

Also due to the redundancies in storage of polar points, the program was optimized so that if the Latitude were North or South 90° (the poles) the algorithm does not run through each Longitude point, as the Station vector will be the same in all 360 cases. It merely makes the calculations once. This way, up to 718 sets of calculations (up to 18 equations per set) are saved for each ten minute increment (called a timestep), or up to 1,912,752 calculations per simulated day. These savings are significant when analyzing a Halo constellation, which is run for approximately 687 simulated days (saving approximately 438,020,208 calculations).

Program Flowchart

A flowchart of CVG begins on the next page. The second page also contains a diagram to help the reader visualize how the vectors are defined in the program at various points.

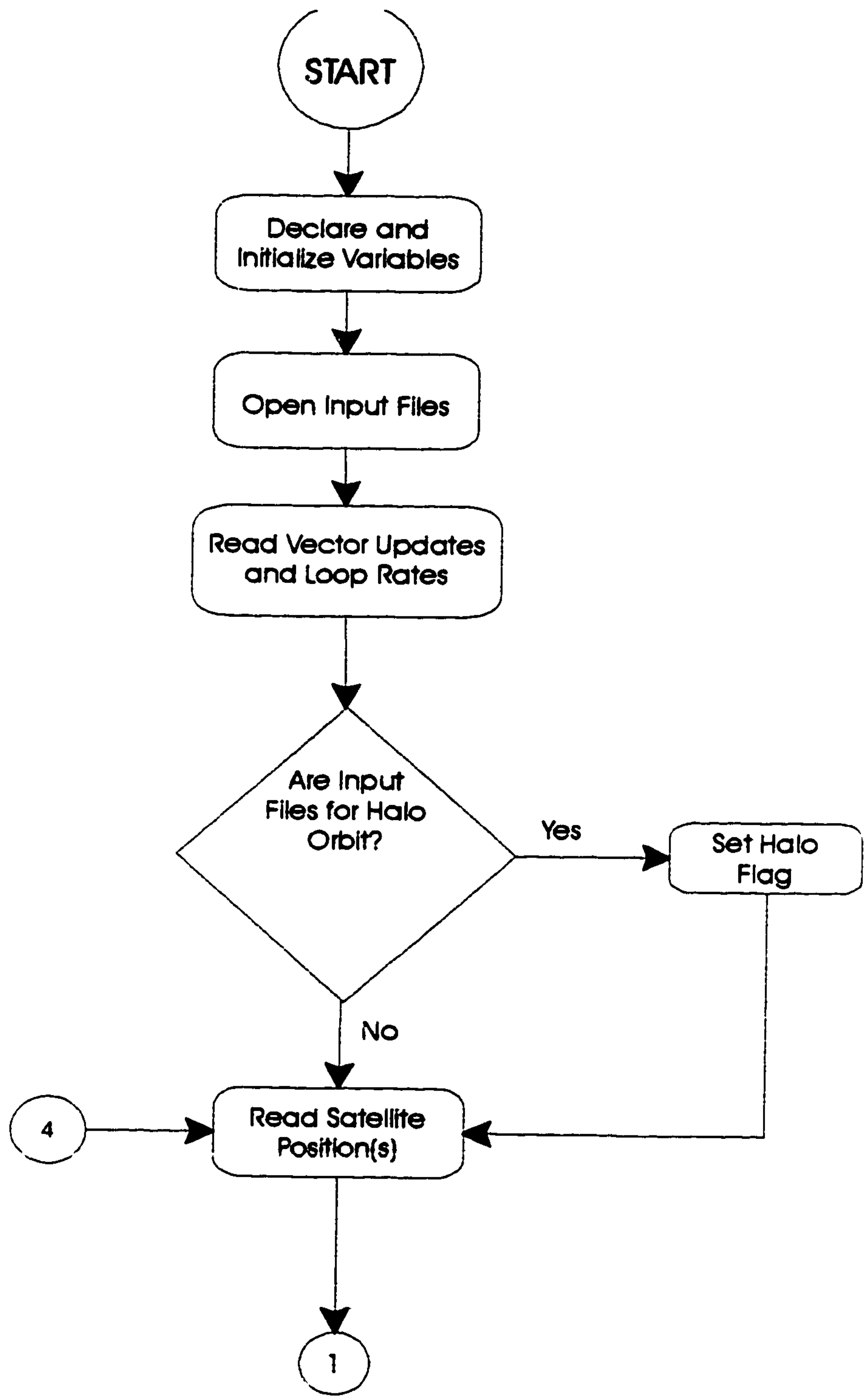

Figure 2 Flowchart of Coverage Analysis Program (CVG)

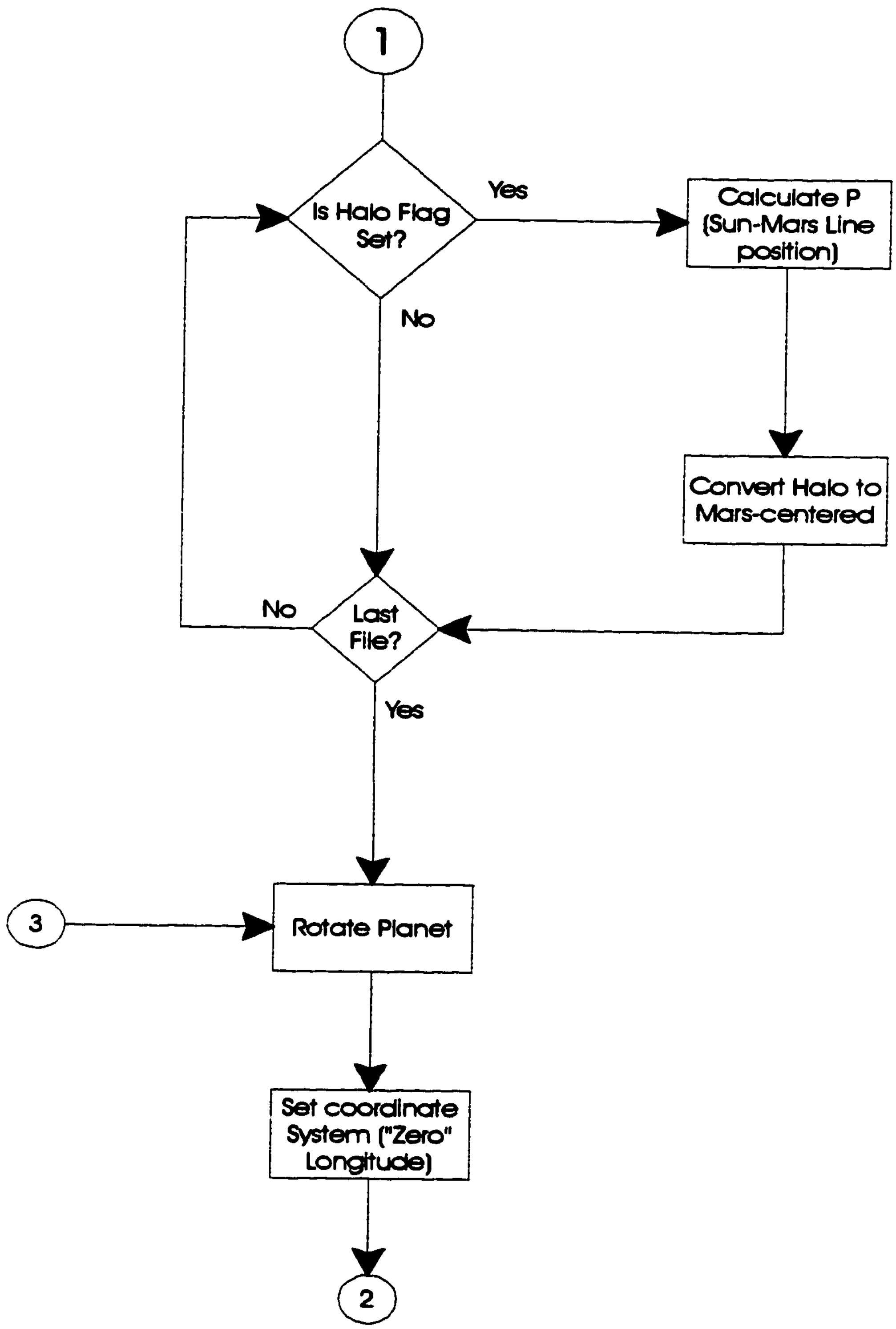

Figure 3 Flowchart of Coverage Analysis Program (CVG) (cont.)

et Ha

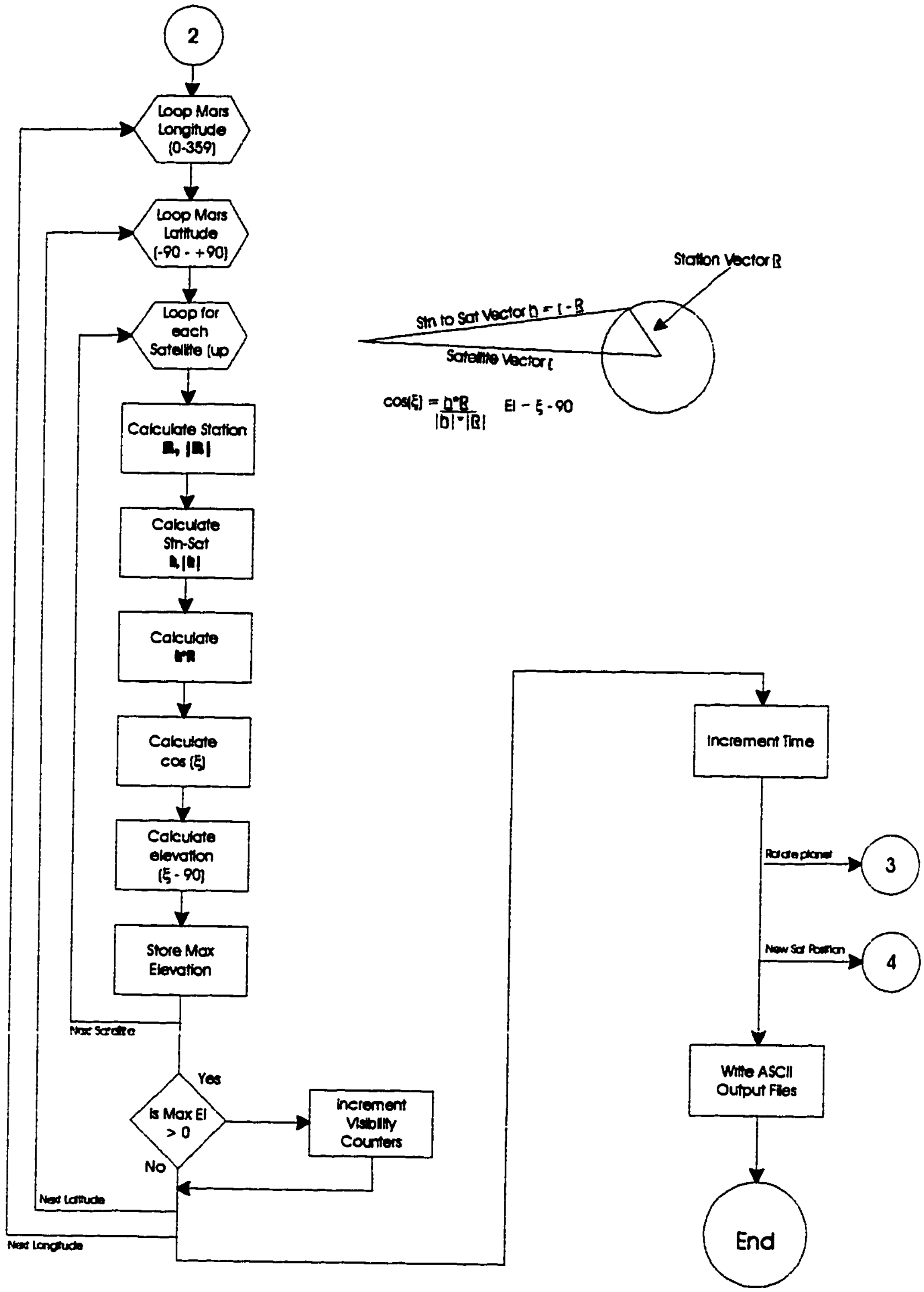

Figure 4 Flowchart of Coverage Analysis Program (CVG) (cont.).

Program Listing

A listing of the source code for the coverage analysis program is included below.

```
/**************************************************************************/
/*          cvg_c.c (v3) - Mark Danehy Jul 1995                          */
/*  A program to determine the communications coverage provided by a     */
/*  satellite constellation at Mars.  The program will determine if      */
/*  any one of a group of satellites is visible at a ground locations    */
/*  around the planet, and determine if the satellite is visible at      */
/*  zero degrees, five degrees, and ten degrees above the horizon.       */
/**************************************************************************/

/**************************************************************************/
/*  IF THE TIME INCREMENT IS CHANGED (NOW 598.93 SEC PER STEP)     */
/* MUST CHANGE STEP_SUN_LAT () DIVISOR FOR LAT (99101.85164 NOW, */
/* #STEPS / MARS YEAR), AND STEP_LONG () LON MULTIPLIER (-2.4324)   */
/**************************************************************************/

#include <stdio.h>
#include <string.h>
#include <stdlib.h>
#include <math.h>

#define OUT1 "zero_c.out"
#define OUT2 "ten_c.out"
#define OUT3 "thirty_c.out"
#define INPUT "input.txt"
#define ERRORFILE "error.txt"
#define PI 3.14159265358979323846
#define RADIUS 3393.
#define STEPS 148
#define DEG_PER_STEP 2.4324
#define EQUATOR_TILT 23.98
#define STEP_YEAR 99101.85164
```

```
void main (void) {

/* variable declarations */
/* filename stores filenames of data files, string holds each line of data, */
/* chr used to take the answer to questions */
char filename[6][15], chr, string[200];
int i, j, k = 0, n, flag, sat_steps, mars_ctr, time_step, time, lat, lon;

/* end_of_file is a flag to signal time to stop reads */
int end_of_file = 0;


/* zero_matrix, five_matrix, and ten_matrix are where the */
/* 0, 5, and 10 deg visibility will be counted memory for */
/* cvg_matrix will be allocated  dynamically */

int  zero_matrix [181] [360];
int  five_matrix [181] [360];
int  ten_matrix [181] [360];

/* out_matrix will hold the percentage visibility */
double out_matrix [181] [360];

/* f_p is file pointer array for input files, out is array for output files */
FILE *f_p[6], *i_f, *e_f;

/* other variables */
double elev, max_el, time1, time2, dummy, ss_lat, ss_long;
double stn_pos[3], stn_sat[3], dum_vec[3];
double vector[6][3];

/* Function prototypes */

void translate (double [][3], int, int);
void write_output (double [][360], char * );
void get_new_sat_posn (FILE * [], int, double [][3], FILE *, int *);
void set_stn_xyz (int, int, double, double, double *);
void calc_stn_sat_vector (double *,double *,double *);
double calc_elev (double *, double *);
void imxinit (int [][360], int);
void mxscale (double [][360], int [][360], int, int, double);
```

```
void vmxmul(double [][3], double [], double []);
void step_long (double *, int);
void step_sun_lat (double *, int);

time = 0;
dummy = 0.;

/* initialize the program matrices */

imxinit( zero_matrix, 0);
imxinit( five_matrix, 0);
imxinit( ten_matrix, 0);


/* open the input and error files */

i_f = fopen(INPUT,"r");
e_f = fopen(ERRORFILE,"w");

/* read the number of datafiles */

fgets(string,25,i_f);
sscanf(string," %d",&n);

/* read the filenames of the datafiles */
/* and open the input files */


flag = 0;
for(i=0; i<n; i++){
   fgets(filename[i],15,i_f);

   /* get rid of CR in the filename string */
   j=strlen(filename[i]);
   filename[i][j-1] = '\0';

   /* open file */
   f_p[i] = fopen(filename[i],"r");
   /* if error, f_p[i] will be NULL. !NULL triggers if statement*/
   if (!f_p[i]) {
            fprintf(e_f,"\nError opening file %s",filename[i]);
```

```
                flag = 1;
                } /* end if */
    }/* end for */

  /* Halo = y/n */

  fgets(string,50,i_f);
  sscanf(string," %c",&chr);

  /* close input file */
  fclose(i_f);


  /* Terminate the program if there was a problem opening the files */

  if (flag) exit(0);


  /* Otherwise, continue */

  if ( (chr == 'n') || (chr == 'N') ) sat_steps = 1, mars_ctr = 1;
  else sat_steps = STEPS, mars_ctr = 0;

  /* get/throw out the first two lines of each file */
  /* (title line) to get to the data */

  for (i = 0; i < n; i++) {

     fgets (string, 200, f_p[i]);
     fgets (string, 200, f_p[i]);

     /* end for i */
     }


  while ( !end_of_file ) {

     /* if the remainder of (time % sat_steps) is zero, then time is a */
     /* integral multiple of sat_steps and it is time to update the sat */
     /* position.  If not zero, [!() is false] then not time yet */
```

```
if ( !(time % sat_steps) ) {

        /* get_new_sat_posn reads the input files.  if at end of file */
        /* g_n_s_p sets end_of_file to 1.  This should allow the loop */
        /* while (!end_of_file) to continue even though the input file */
        /* is at eof. Previous test (!feof) did not allow the loop to */
        /* continue */

        get_new_sat_posn ( f_p, n, vector, e_f, &end_of_file);

        if (!end_of_file && !mars_ctr) translate (vector, time, n);

        /* end if */
        }

if (!end_of_file) {

        /*  step_long rotates the planet about it's axis,  */
        /*  modeling the longitude of the sun as mars goes */
        /*  through it's day.  step_sun_lat models tilt of */
        /*  mars for calc_stn_xyz.                         */

        step_sun_lat (&ss_lat, time);
        step_long (&ss_long, time);

        /* check every one_degree by one_degree point for visibility */

        for (lat = 0; lat <= 180; lat++) {

                for (lon = 0; lon <= 359; lon++) {

                        /* if lat = 0,180 then at pole, and only need   */
                        /* one calculation, vice every degree by degree */
                        /* so set lon[gitude] to 180, calc elevation    */
                        /* and update the entire matrix row             */

                        if ( lat == 0 || lat == 180 ) lon = 180;

                        /* convert stn lat/long to  Mars_ctr_inertial coord */
                        set_stn_xyz (lat, lon, ss_lat, ss_long, stn_pos);
```

```
/* set/reset max_el to zero */
max_el = 0.;

/* calculate stn to sat vectors, and calc elevation */

for (i = 0; i < n; i++) {

        dum_vec[0] = vector[i][0];
        dum_vec[1] = vector[i][1];
        dum_vec[2] = vector[i][2];

        calc_stn_sat_vector (stn_pos, dum_vec, stn_sat);

        elev = calc_elev (stn_pos, stn_sat);

        /* elev returned in degrees */
        if (elev > max_el) max_el = elev;

        /* end for i*/
        }

if (max_el > 0.) {

        if (lat == 0 || lat == 180)
                for (i = 0; i <= 359; i++) zero_matrix[lat][i]
++;

        else zero_matrix[lat][lon] ++;

        if (max_el > 10.) {

                if (lat == 0 || lat == 180)
                        for (i = 0; i <= 359; i++)
five_matrix[lat][i] ++;

                else five_matrix[lat][lon] ++;

                if (max_el > 30.) {
                        if (lat == 0 || lat == 180)
                                for (i = 0; i <= 359; i++)
ten_matrix[lat][i] ++;
```

```
                                        else ten_matrix[lat][lon] ++;

                                        /* end if max_el > 30 */
                                        }


                                    /* end if max_el > 10 */
                                    }

                                /* end if max_el > 0 */
                                }

                            /* set lon to 359 (end of the row) if at pole */

                            if ( lat == 0 || lat == 180 ) lon = 359;

                            /* end for lon */
                            }


                        /* end for lat */
                        }


                    /* increment time */
                    time++;

                    /* end if !end_of_file */
                    }


            /* end while !feof f_p[0]*/
            }


        /* take each element of cvg_matrix and divide by time. this gives */
        /* percentage time that satellite has been visible at that point   */

        /* use dummy to hold inverse of time, used to scale matrices */

        dummy = 1./ (double) time;
```

```
for (i = 0; i < 3; i++){

   switch (i) {

              case 0:

                     /* scale zero_matrix by inverse of time */

                     mxscale ( out_matrix, zero_matrix, 181, 360, dummy);

                     /* set two of the corner values  of out_matrix to */
                     /* 1.0 and 0.0, to give output right color scaling */

                     out_matrix[0][0] = 0.0;
                     out_matrix[0][1] = 1.0;

                     /* write the output (out_matrix) to file) */

                     write_output (out_matrix, OUT1);

                     break;

             case 1:

                     /* scale five_matrix by inverse of time */

                     mxscale ( out_matrix, five_matrix, 181, 360, dummy);

                     /* set two of the corner values  of out_matrix to */
                     /* 1.0 and 0.0, to give output right color scaling */

                     out_matrix[0][0] = 0.0;
                     out_matrix[0][1] = 1.0;

                     /* write the output (out_matrix) to file) */
                     write_output (out_matrix, OUT2);

                     break;
```

```
                case 2:

                        /* scale ten_matrix by inverse of time */

                        mxscale (out_matrix, ten_matrix, 181, 360, dummy);
                        /* set two of the corner values  of out_matrix to */
                        /* 1.0 and 0.0, to give output right color scaling */

                        out_matrix[0][0] = 0.0;
                        out_matrix[0][1] = 1.0;

                        /* write the output (out_matrix) to file) */

                        write_output (out_matrix, OUT3);

                        break;

                /* end switch i */
                }

    /* end for i */
    }

/* end main */

fclose(e_f);
return;
}



void write_output (double m[][360], char *string ) {

int i, j;
FILE *fptr;

/* open output file */

fptr = fopen (string,"w");
```

```
for (i = 0; i < 360; i++) {


    for (j = 180; j >= 0; j--) fprintf (fptr, "%5.3lf,", m[j][i]);

    fprintf (fptr,"\n");


    /* end for i */
    }

/* close output file */

fclose(fptr);

return;
/* end function */
}



void translate( double vec[][3], int time, int n )

{

double m1[3][3];
double v1[3], v2[3];
double theta, N, psi_o, psi, r, mu;
int i;

/* function declaration */
void vmxmul ( double [][3], double [], double [] );

/* r and mu are Halo/Rotating Frame constants for Sun/Mars */

r = 2.2794e8;
mu = 3.239e-7;
psi_o = 0.;

/* N is the mean motion of Mars about the Sun expressed in radians */
/* per timestep [2*pi/(686.98 * 86400 secs/day) * 598.93 secs/step */
```

```
N = 6.340129075e-5;

/* theta = tilt of mars */

theta = 25.19 * PI/180.;

/* psi = angle mars has traveled about the sun */

psi = psi_o + N * (double)time;

/* set up rotation matrix */

m1[0][0] = cos (psi);
m1[0][1] = - sin (psi);
m1[0][2] = 0.;

m1[1][0] = cos (theta) * sin (psi);
m1[1][1] = cos (theta) * cos (psi);
m1[1][2] = sin (theta);

m1[2][0] = -sin (theta) * sin (psi);
m1[2][1] = -sin (theta) * cos (psi);
m1[2][2] = cos (theta);


for (i = 0; i < n; i++){

    /* translate from barycentric to mars_centered */

  v1[0] = (vec[i][0] - (1 - mu) ) * r;
  v1[1] = vec[i][1] * r;
  v1[2] = vec[i][2] * r;

  /* de-spin coordinate system to mars centered */
    /* v1 is mars centered rotating frame */

  vmxmul (m1, v1, v2);

    /* v2 is now mars centered inertial */

  vec[i][0] = v2[0];
```

```
    vec[i][1] = v2[1];
    vec[i][2] = v2[2];

    /* end for i */
    }

return;
/* end function */
}




void get_new_sat_posn ( FILE * f_p[], int n, double vector[][3], FILE *e_f , int *eof)

{

int i;
char string[200];
double dummy, time1, time2;


if ( feof( f_p[0] ) ) *eof = 1;

else {

   /* read input files and get new satellite vectors */

   for (i = 0; i < n; i++) {

              fgets (string, 200, f_p[i]);

              /* dummy's in sscanf statement are to allow the read of all the */
              /* variables we want without allocating too many variables */

              sscanf(string, " %lf %lf %lf %lf %lf %lf %lf %lf %lf", &dummy, \
                     &time2, &dummy, &vector[i][0], &vector[i][1], \
                     &vector[i][2], &dummy, &dummy, &dummy);
```

```
                /* if i=0, then this is first file. want to save time to check */
                /* against other files.  should all match */


                if (!i) time1 = time2;
                else if (time2 != time1) {

                        fprintf (e_f,"\n\nError - Sat position times do ");
                        fprintf(e_f,"not match. Check input files");

                        exit (1);
                        /* end if time2 != time1 */
                        }

                /* end for */
                }

    /* end else */
    }

return;
/* end function */
}


void step_sun_lat (double *lat, int time) {


/* assumes that mars travels in a circular orbit about sun.    */
/* lat = Mars tilt * sin function with 598.93 sec step and     */
/* period of 1 Mars year.  STEP_YEAR is # timesteps per Mars */
/* year (686.98 Earth Days per Mars year * 86400 sec per Earth */
/* day / 598.93 sec per timestep                          */

*lat = EQUATOR_TILT * sin ( 2. * PI * (double) time / STEP_YEAR);

return;

/* end function */
```

```
}

void step_long (double *lon, int time) {

/* Mars rotates 2.4324 degrees per time step. % operator */
/* return the remainder of time / 148 (resets to zero    */
/* when time hits multiple of 148 )                      */

*lon = -DEG_PER_STEP * (time % STEPS);

return;

/* end function */
}




void set_stn_xyz (int lat, int lon, double sun_lat, double sun_lon, double * pos)
{

/* function to convert station lat/long to xyz coords */
double r = RADIUS ;

/* s_lon, s_lat are station lat/long corrected in Mars_centered */
/* inertial (lat - 90) converts loop counter to latitude */
/* sun_lat and sun_lon are used to correct for the tilt of M */

double s_lat = (double)(lat - 90) - sun_lat;
double s_lon = (double)lon - sun_lon;

/* convert s_lat and s_lon to radians from degrees */
s_lat *= (PI/180.);
s_lon *= (PI/180.);

pos[0] = r * cos( s_lat ) * cos( s_lon );
pos[1] = r * cos( s_lat ) * sin( s_lon );
pos[2] = r * sin( s_lat );
```

```
return;
/* end function */
}




void calc_stn_sat_vector ( double * stn, double * sat, double * stn_sat)
{
/* calculate the vector from the surface point to the satellite, */
/* when both are specified in planet-centered coordinates


/* debugging

printf("\nstn = %f, %f, %f",stn[0], stn[1], stn[2]);
printf("\nsat = %f, %f, %f",sat[0], sat[1], sat[2]);   */

stn_sat[0] = sat[0] - stn[0];
stn_sat[1] = sat[1] - stn[1];
stn_sat[2] = sat[2] - stn[2];

/* debugging
printf("\nstn_sat = %f, %f, %f",stn_sat[0], stn_sat[1], stn_sat[2]); */

return;
/* end function */
}




double calc_elev (double * stn,  double * stn_sat)
{

/* Performs calculations for elevation in rads. */
/* Last step converts rads to degrees */
```

```
double elev, mag1, mag2, dot, cos_theta, theta;
double v1[3];

/* function declarations */
double vmag (double [], int );
double vdot (double [], double []);

/* invert stn for the dot product */

v1[0] = -stn[0];
v1[1] = -stn[1];
v1[2] = -stn[2];


/* get magnitude of the two vectors */
mag1 = vmag(stn, 3);
mag2 = vmag(stn_sat, 3);

/* debugging
printf("\nmag1 = %f, mag2 = %f", mag1, mag2); */


/* calculate dot product, uses v1 (inverse */
/* of stn) to get the proper angle        */

dot = vdot (v1, stn_sat);

/* debugging
printf("\ndot = %f", dot);  */


/* cos theta = dot product divided by */
/* product of magnitudes */

cos_theta = dot / (mag1 * mag2);

/* debugging
printf("\ncos(theta) = %f", cos_theta); */

/* find the arc cos of (cos_theta) to get theta */
theta = acos (cos_theta);
```

```
/* debugging
printf("\ntheta = %f", theta); */


/* a zero degree elevation corresponds to a 90 degree       */
/* angle between the station vector and the station-to-sat */
/* vector. to find the elevation, subtract 90 from the     */
/* elevation (PI/2 because elev is in radians at this point */

elev = theta - (PI/2.);

/* convert elev from rads to degrees */

elev *= 180./PI;

/* debugging
printf("\nelev = %f", elev);  */


return elev;
/* end function */
}




double vmag (double *v, int n)

{

/* calculate the magnitude of the n-vector v */

int i;
double mag = 0., temp = 0.;


for(i = 0; i < n; i++) temp += pow ( v[i], 2.);

mag = sqrt (temp);
```

```
return mag;

/* end function */
}



void imxinit (int m[][360], int val)
{

/* initialize integer matrix[181][360] */

int i, j;

for (i = 0; i < 181; i++){

    for (j = 0; j < 360; j++) m[i][j] = val;

    /* end for i */
    }

return;
/* end function */
}



double vdot ( double *a, double *b)

{

int i;
double dot = 0.;


if ( (a == NULL) || (b == NULL) ) {

    printf("\nInvalid vector passed to vdot");
    return 0.0;
```

```
   /* end if */
   }


for (i = 0; i < 3; i++) dot += a[i] * b[i];


return dot;
/* edn function */
}




void vmxmul(double m[][3], double v[], double y[])
{

double sum;
int i,j;

for (i = 0; i < 3; i++) {

  /* for each row of m[][] */
  sum = 0.0;

  for (j = 0; j < 3; j++) {

              /* for each col of m[][] */
              sum += m[i][j] * v[j];

              /* end for j */
              }

  y[i] = sum;

  /* end for i */
  }
```

```
return;
/* end function */
}

void mxscale (double fm[][360], int im[][360], int rows, int cols, double val) {

int i,j;

for (i = 0; i < rows; i++) {

   for (j = 0; j < cols; j++) fm[i][j] = (double) im[i][j] * val;

   /* end for i */
   }

return;
/* end function */
}
```

APPENDIX C
Glossary

Apoapsis	Highest (farthest from the planet's center of mass) point in a satellite's orbit (for Earth orbiting satellites this would be called "apogee").
Constellation	The group of communications satellites.
Constellation Availability	Constellation Availability is the percent time that at least one satellite in the constellation is visible to the "user."
CR3BP	Circular, Restricted Three Body Problem - set of simplifying assumptions made to the Three Body Problem to aid in determination of solutions.
Periapsis	Lowest (closest to the planet's center of mass) point in a satellite's orbit (for Earth orbiting satellites this would be called "perigee").
Satellite Vector	The vector from the center of the planet to the satellite.
Station Vector	The vector from the center of the planet to the user.
Station-Satellite Vector	The vector from the ground observer to the satellite, equal to the vector difference between the Satellite Vector and the Station Vector.
Timestep	One time interval used in the CVG program. Approximately equal to ten minutes in this thesis.
User	Any system wishing to make use of the Earth - Mars Network.

APPENDIX D
Modifications to Draim Orbit for Mars Constellation

In his paper on the topic[16], retired Navy Captain John Draim proposed a satellite constellation which would provide continuous coverage of Earth, using only four satellites. Capt Draim proposed placing the satellites in inclined, elliptical orbits, and setting apogee for two of the satellites in the Northern Hemisphere, and two in the Southern. By properly configuring the orbit planes and the satellites in them, complete geometric coverage of the planet, at zero degrees elevation, is achieved.

In his paper, Capt Drain described the process which was used to determine the proper semi-major axis to use with the orbit. From the description, he used geometric arguments, testing various configurations by trial and error. No formulas were presented to assist one in determining these parameters independently. Nonetheless, the author was still required to determine proper orbital parameters for the planet Mars.

Given that Capt Draim used a geometric argument in his determination of the Earth constellation, the author chose a similar path to determine the Mars parameters. After determining the semi-major axis of the Earth constellation orbits, the semi-major axis of the Mars orbits was determined by calculating the ratio of the Earth semi-major

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[16]Draim, J.E., A Common-Period Four-Satellite Continuous Global Coverage Constellation, *Journal of Guidance, Control, and Dynamics*, Vol. 10, No. 5, Sept-Oct 1987, 492-499

axis to the radius of the Earth, then using that ratio to determine the semi-major axis of the Mars orbit. The other orbital parameters for the constellation were taken directly from the Draim paper.

These orbital parameters were then input to an orbit propagator, the program POHOP from the Jet Propulsion Laboratory. The output of POHOP was then input into the CVG program, written by the author. CVG provides a 181x360 matrix detailing the percentage time that any one satellite in the constellation is visible from points on the surface of Mars, corresponding to degree by degree latitude/longitude intersections (67N, 132W; 04S, 72E; etc.). As a control, the Draim orbit around Earth was checked by CVG, and found to provide full geometric (zero degree elevation or higher) coverage of Earth.

Running CVG with the scaled Mars constellation numbers, CVG reported that this constellation did not provide full geometric coverage of Mars. The author believes that this is caused by the mass parameters of Mars as opposed to Earth. Since decreasing the semi-major axis would only reduce the constellation's performance, the author increased it, and repeated the coverage determination process. The result still did not meet the requirement of 100% geometric coverage, and the process was repeated. After several attempts, an orbit size was found which provided 100% coverage. It's period was slightly over two sols (Mars days, approx 24.6 hr). Given that this was a communications satellite, the simpler the orbit configuration, the better. The author decided that an orbital period of a whole number of sols was the best choice. This would provide the user a ground track which repeated. Using these criteria, the orbital period was chosen to be three sols (approximately 74 hours), which was the next whole sol after 100% coverage was achieved.

The final orbital parameters used are illustrated in Table D-1.

Table D-1 - Comparison of Draim Orbit Parameters			
Parameter	Draim Original	Mars-Modified Draim	Final Mars Draim
semi-major axis	45120 km	23910 km	42560 km
eccentricity	0.263	0.263	0.263
inclination	40.0 deg	40.0 deg	40.0 deg
longitude of ascending node	0.0 deg	0.0 deg	0.0 deg
argument of periapsis	-90.0 deg	-90.0 deg	-90.0 deg
mean anomaly	0.0 deg	0.0 deg	0.0 deg

APPENDIX E

Summary of Spacecraft Configurations for each Scenario

This appendix provides a discussion of fault tolerance, with regard to the constellation, and a concise summary of spacecraft configurations for each scenario.

Fault Tolerance

In constellation design, fault tolerance is a matter of designing the constellation to be able to tolerate and recover from failures. In this case, that would mean having multiple spacecraft, any of which could perform critical mission functions.

This discussion will purposely not include a discussion of the fault tolerance of individual spacecraft. The design of individual spacecraft will necessarily depend upon the constellation design, how much redundancy is built into it, the customer's budget, and the customer's willingness to accept risk. Design features like multiple solid state communications amplifiers, as opposed to a single Traveling Wave Tube (TWT) amplifier, would allow a graceful degradation of transmitter power as amplifiers fail.

No analysis has been accomplished to determine the degree of fault tolerance these constellation designs provide. This area would likely be a good topic for future work, removing one satellite at a time from the CVG program runs, and analyzing the results.

Intuitively, the most fault tolerant are the Low and Medium Orbit Relay Satellite configurations. The six-satellite configurations would seem to provide the most redundancy. The four-satellite Aerosynchronous configuration would also provide a means for graceful degradation (the four satellites are all in the same orbital plane, and could be easily rephased to assume a three satellite configuration.

The least fault tolerant of these constellation configurations would be the Halo and the CPI configurations. These constellations are critically populated, with no fallback configuration. The only option in these cases is to lose the coverage over the affected region. The loss of one Halo Orbiter, for instance, would cause an outage period of about twelve hours on either the day or night side of the planet. The only option would be to store data during the blackout period, and uplink it once the lander moves back into communications coverage.

Halo Orbiting Relay Satellite

The Halo Orbiter is located approximately one million kilometers from Mars, and thus requires a large, focused antenna to keep a majority of the signal illuminating Mars (and not wasting it radiating the space around the planet). To aid in keeping the signal as focused as possible on Mars, this link uses a very high transmitter frequency, which is more easily focused and wastes less RF signal energy.

Earth-link Antenna: 5 m parabolic.

Earth-link Transmitter Power: 30 W (14.8 dBW or 44.8 dBm).

Earth-link Transmitter Frequency: 30 GHz.

Mars-link Antenna: 1.8 m parabolic (based upon the angular size of Mars at one million kilometers distance.

Mars-link Transmitter Power: 20 Watts (13 dBW or 43 dBm).

Mars-link Transmitter Frequency: 30 GHz.

Configuration Advantages

The Relay "sees" nearly half or the planet from this vantage point. The Relay is visible to the ground user for a large portion of the day/night, with coverage gaps lasting approximately fifteen minutes.

The Sun is always visible to the Relay satellite, so large batteries to carry eclipse loads are not required (lower capacity batteries, to maintain critical spacecraft functions during attitude anomalies, such as tumbling, are still required).

Only two satellites are required to make this configuration fully operational.

Configuration Disadvantages

A large Mars-pointing antenna and high transmitter frequency are required to overcome the losses induced by the great distance from the relay to Mars. The large antenna makes solar panel mounting and shadowing a more difficult problem, and the high transmitter frequency makes the design of lander communications systems more difficult. The fact that this configuration uses two Relay satellites makes it especially vulnerable to a satellite failure (if one satellite fails, half the planet loses communications coverage for approximately 12 hour intervals).

Common Period, Inclined Relay Satellite

Earth-link Antenna: 5 m parabolic.

Earth-link Transmitter Power: 30 W (14.8 dBW or 44.8 dBm).

Earth-link Transmitter Frequency: 30 GHz.

Mars-link Antenna:

Operational Period: Horn antenna (based upon the angular size of Mars at apoapsis) providing full Mars coverage.

Non-operational Period: Helix (based on the angular size near periapsis).

Mars-link Transmitter Power: 4 Watts (6 dBW or 36 dBm).

Mars-link Transmitter Frequency: 2 GHz (due to constraints imposed by using Helix antennas during non-operational period).

Configuration Advantages

Four satellite constellation will likely provide a better degradation scenario than the Halo configuration, though no work has been done to compare the degradation characteristics of individual constellations. This constellation actually provides the best communications coverage of the surface, though it uses twice the satellites of the Halo configuration to do so.

Configuration Disadvantages

The disadvantage of this configuration is that the Halo configuration does nearly the same job with half the satellites. Satellite tracking for this constellation will be difficult, as one satellite will have to be tracked, then the ground antenna must be switched to the next satellite. Where with the Halo case, ground stations can be set to point at one spot in space and track that one spot (as current Earth telescope clock-drives do), CPI ground stations must track a north or south elliptical path across the sky.

Aerosynchronous Relay Satellite

Earth-link Antenna: 5 m parabolic.

Earth-link Transmitter Power: 30 W (14.8 dBW or 44.8 dBm).

Earth-link Transmitter Frequency: 30 GHz.

Mars-link Antenna: Horn (based upon the angular width of Mars at aerosynchronous altitude.

Mars-link Transmitter Power: 4 Watts (6 dBW or 36 dBm).

Mars-link Transmitter Frequency: 15 GHz.

Configuration Advantages

Four satellite configuration is highly fault tolerant, being able to rephase to a three satellite configuration, causing only some communication gaps. Three satellite configuration does not have this ability, as three satellites are the minimum required to provide coverage from this altitude.

Ground antenna pointing is greatly simplified, as the satellite remains essentially in one place in the sky (no or minimal tracking is required).

Configuration Disadvantages

No polar coverage provided by this constellation (constellation is unacceptable for applications requiring polar coverage, such as MESUR). Areas in coverage gaps tend to remain in gaps, there is little travel of the holes to equalize the coverage.

Low and Medium Orbiting Relay Satellite

Earth-link Antenna: 5 m parabolic.

Earth-link Transmitter Power: 30 W (14.8 dBW or 44.8 dBm).

Earth-link Transmitter Frequency: 30 GHz.

Mars-link Antenna: Helix (based upon the angular width of Mars from low altitude).

Mars-link Transmitter Power: 4 Watts (6 dBW or 36 dBm).

Mars-link Transmitter Frequency: 400 MHz (based upon parameter of Helix antenna and signal dispersal).

Configuration Advantages

One of the most fault tolerant constellation designs, as every bit of the planet is covered by each satellite. A loss of one satellite would most probably just increase the revisit time of any spacecraft to a particular ground user, requiring more data be stored aboard the ground platform, and more data be read out during the subsequent satellite overflight.

Configuration Disadvantages

Lowest communications coverage numbers of all constellations examined, due to low Field of View from this altitude. Most complex Earth-communications switching of all configurations examined.